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F-1 PRESSURIZATION SYSTEM ANALYSIS (U)

2095834 ROCKETDYNE

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NASA Contract NASw-16

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FORM R 18-G-1 (PLATE)

FOREWORD

This report was prepared to present supplementary technical material on the study of propellant tank pressurization systems for use with the F-1 rocket engine. The study was accomplished in accordance with NASA contract NASw-16.

ABSTRACT

A study of the most promising systems for propellant tank pressurization is presented. Reliability, cost analysis, and evaluation of system weights are among the major considerations.

(Unclassified Abstract)

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INTRODUCTION

Rocketdyne, in accordance with NASA contract NASw-16, has conducted a study of propellant tank pressurization systems for use with the F-1 engine. Report R-1559, issued in May 1959, presented the results of the initial phase of this study.

In view of the subsequent deletion of storable propellant capability from the F-1 program, the probability of using liquid hydrogen on upper stages, and acquisition of considerable information concerning liquid helium pressurization systems, it was deemed appropriate to conduct a further study of the problem. This study was limited to the most promising systems. The scope was expanded relative to the initial work to include reliability and cost considerations, as well as an evaluation of systems weights. This report presents the results of this study.

SUMMARY

An analysis was made of eight pressurization systems for the F-1 engine. The systems have been numbered 1 through 8 for identification. The basic characteristics of each system are summarized in a numerical list on page 5.

Factors considered in the analysis were system weight, reliability, facility costs, hardware costs, pressurant costs, operating costs, and state of the art. Considerable study was devoted to determining the most appropriate methods of calculating pressurant requirements.

The numerical results of the study are relative, and valid only for the purpose of comparing the systems. More detailed studies would be required in order to establish the actual configuration, weight, reliability and cost of any one system.

BASIC CHARACTERISTICS, F-1 PRESSURIZATION SYSTEMS

System Number	Fuel Tank Pressurant	Oxidizer Tank Pressurant	Remarks
1	Liquid Helium (LHe)	Liquid Helium	
2	Liquid Nitrogen (LN_2)	Liquid Nitrogen	
3	Liquid Helium or Liquid Nitrogen	Liquid Helium or Liquid Nitrogen	"Compatible System" - Either LN_2 or LHe may be used without hardware changes.
4	Helium Gas (GHe)	Helium Gas	Helium tank cooled with LN_2
5	Helium Gas	Helium Gas	Helium tank cooled with Liquid Hydrogen (LH_2)
6	Liquid Nitrogen	Liquid Oxygen	
7	Liquid Hydrogen	Liquid Oxygen	
8	Helium Gas	Liquid Oxygen	Helium tank cooled with Liquid nitrogen

CONCLUSIONS AND RECOMMENDATIONS

Vehicle reliability and payload weight are the primary factors on which the conclusions and recommendations are based (See Fig. 7). In the F-1 engine development program, reliability is the most important consideration. Therefore, it is recommended that the liquid oxygen/liquid nitrogen pressurization system be selected.

It is further recommended that, after the hydrogen technology has been developed to a high degree of reliability on the J-2 upper stage engine program, the liquid hydrogen system be substituted for the liquid nitrogen system. This change would increase the payload capability by 900 lb per engine.

RELIABILITY AS PRIME CONSIDERATION

Considering system reliability as the most important single factor, the liquid oxygen/liquid nitrogen pressurization system is recommended.

The optimum liquid nitrogen (No. 2) system and the LOX/LN_2 (No. 6) system have the highest, and essentially the same, estimated reliability. The LOX/LN_2 system offers the advantage of approximately 60 to 95 lb payload capability per engine over the optimum liquid nitrogen system. The use of LOX rather than nitrogen for LOX tank pressurization also eliminates the problem of nitrogen absorption by the LOX and reduces the pressurization gas tank volume to approximately one-half. Based on these factors, the LOX/LN_2 system is selected.

PAYLOAD WEIGHT AS PRIME CONSIDERATION

Considering payload weight as the most important single factor, the LOX/LH_2 (No. 7) pressurization system is recommended. The two systems having the lowest estimated weights are the LOX/LH_2 system and the optimum liquid helium system. The LOX/LH_2 system provides 27 lb (0.02%) less payload capacity than the optimum liquid helium system. Since this difference is within estimating accuracy, other factors were examined. Hydrogen has better heat transfer characteristics, is the easier to handle in the liquid state, more readily storable in the liquid state, and offers a higher system reliability. Therefore, the LOX/LH_2 system is selected.

SYSTEM DESCRIPTIONS

In all systems (Fig. 1 through 6) the pressurants are stored in suitable tanks. The pressurant flows from the tank(s) through a heat exchanger, where it is vaporized and/or superheated and then flows into the main propellant tanks.

All systems utilize propellant tank pressure regulators, since this results in minimum pressurant consumption. In the case of Systems 2, 6, and 7 and for the LOX side of 8, it would be practical to use an orifice to control pressurant flow rate and depend on the main tank relief valves to vent any excess pressurant. This would increase the reliability of these systems at the cost of increased system weight.

In Systems 1 and 3, the pressurant gas is loaded into a supply tank in the liquid state and at ambient pressure. The supply tank is fitted with an internal heat exchanger. A controller sensing supply tank pressure positions a bypass valve which diverts a portion of the pressurant from the outlet of the main heat exchanger through the supply tank heat exchanger. This causes supply tank pressure to increase to and be maintained at the controller set-point. The set-point is sufficient to create the required pressure differential for transfer of pressurant into the propellant tanks. (Since the set-point is above the critical temperature and pressure of helium, the pressurant is not actually in the liquid state, and the term "near liquid" has sometimes been applied.)

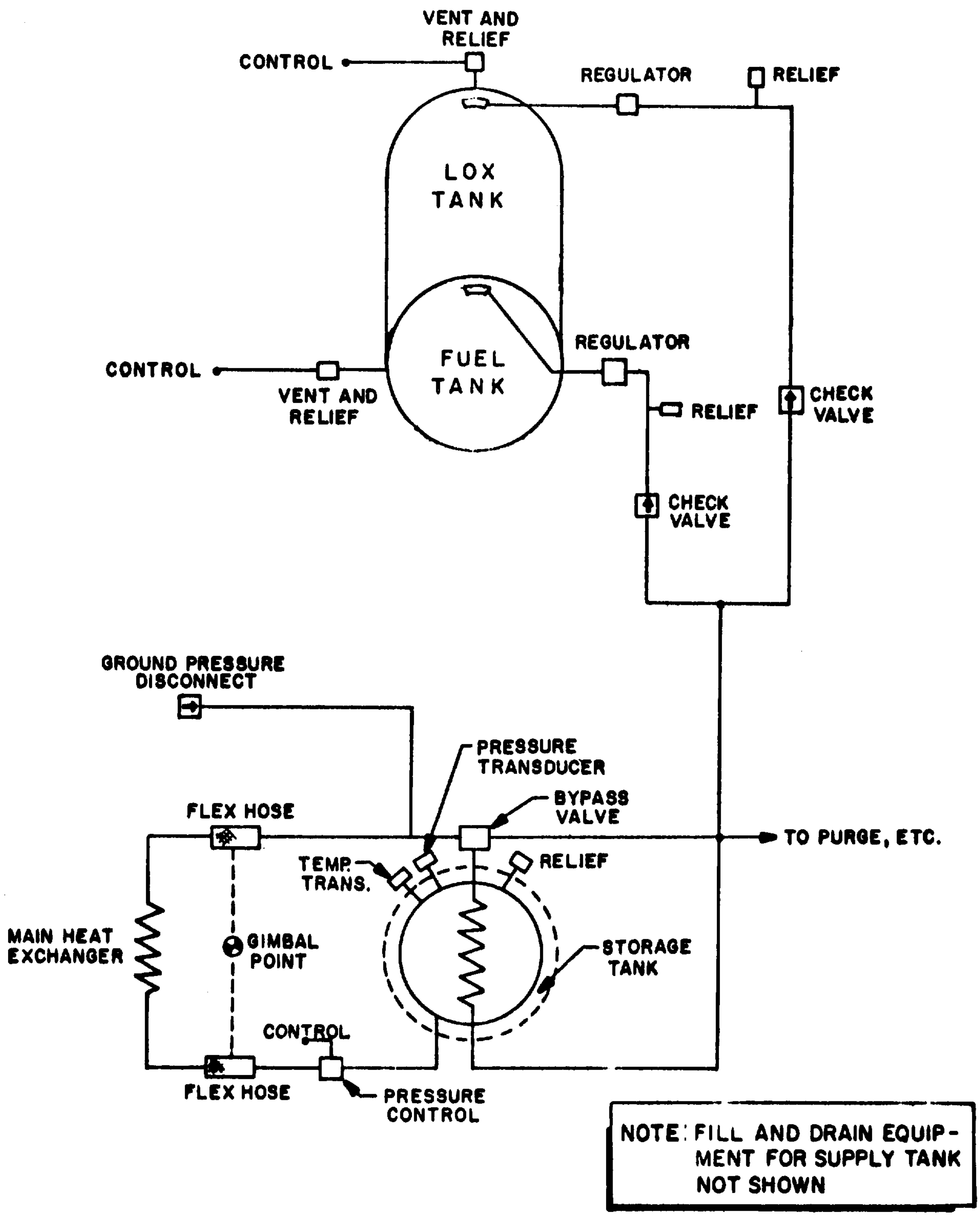


Figure 1. Optimum Near Liquid Helium and Compatible LHe/LN_2 Systems, Systems 1 and 3

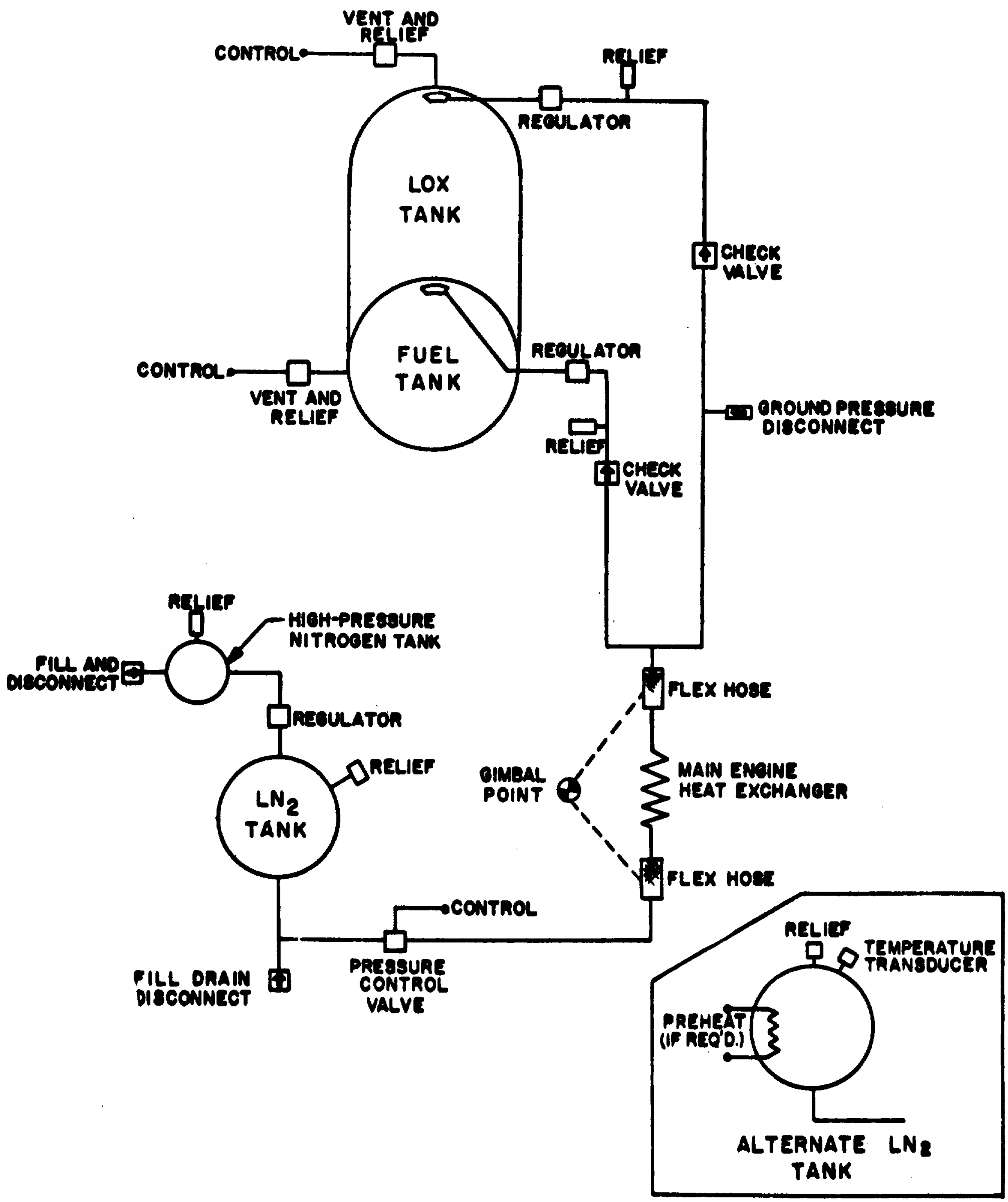


Figure 2. Optimum LN_2 System, System 2

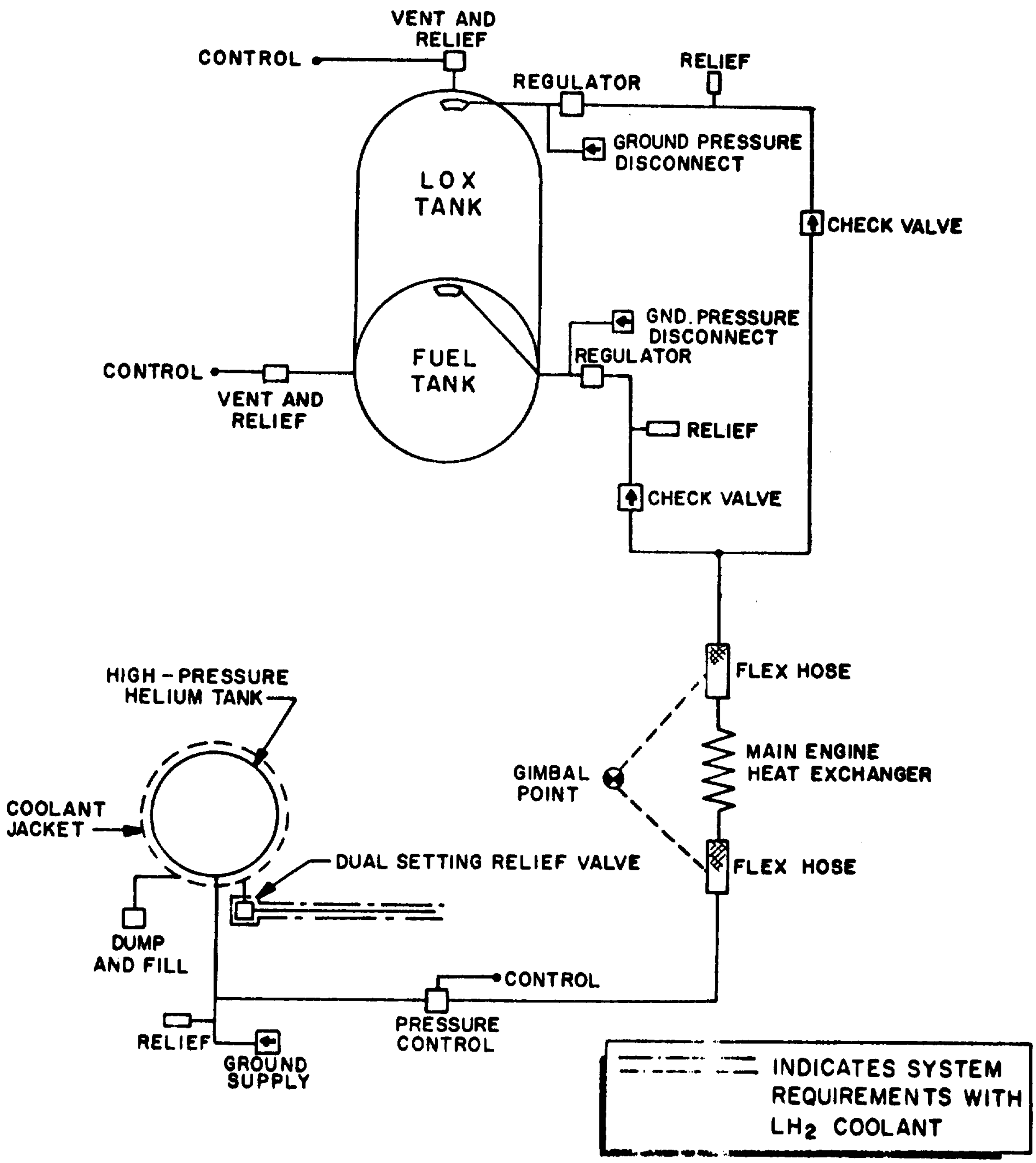


Figure 3. High Pressure Helium System with LN_2 or LH_2 Coolant, Systems 4 and 5

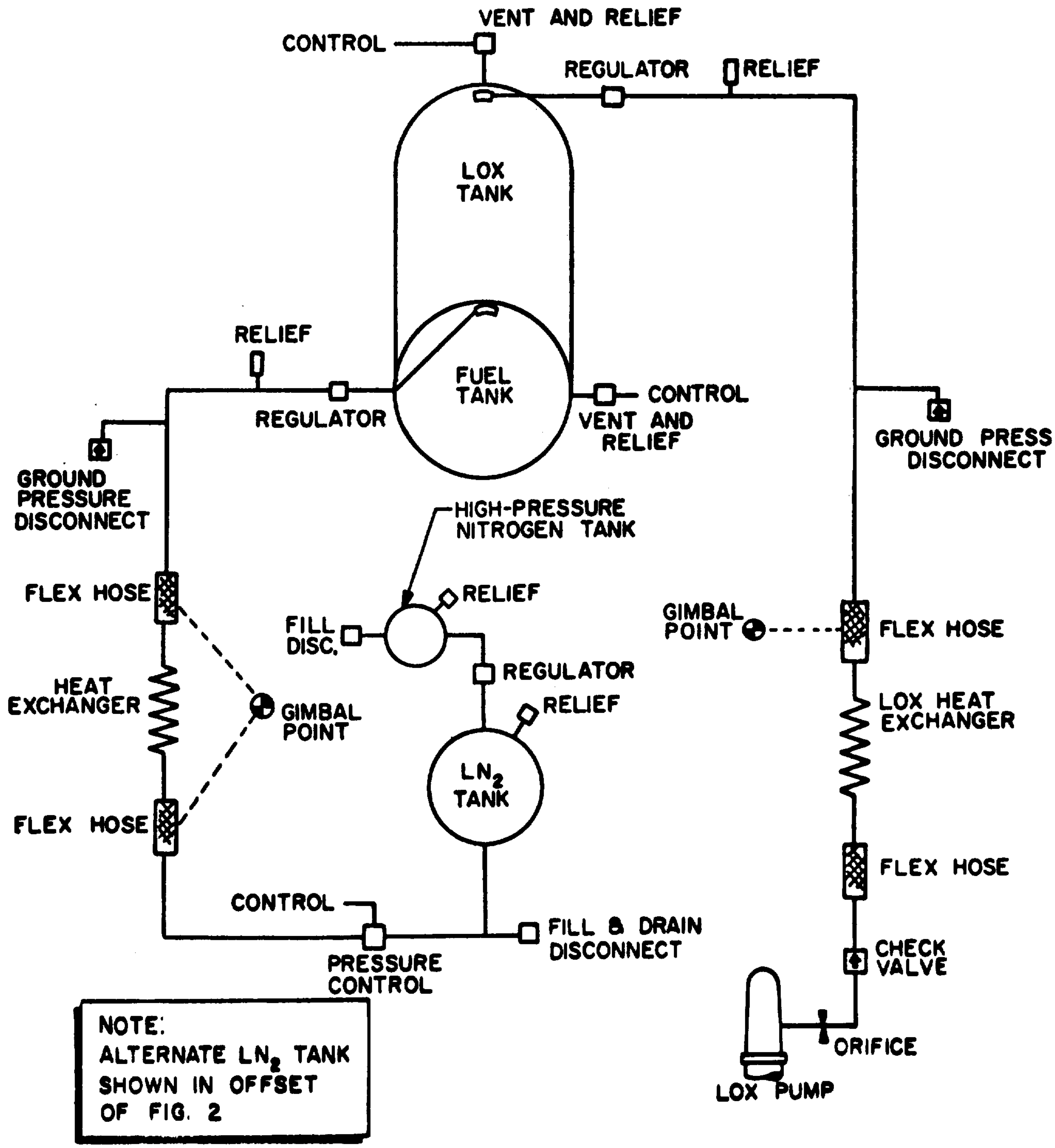


Figure 4. GOX/LN_2 System, System 6

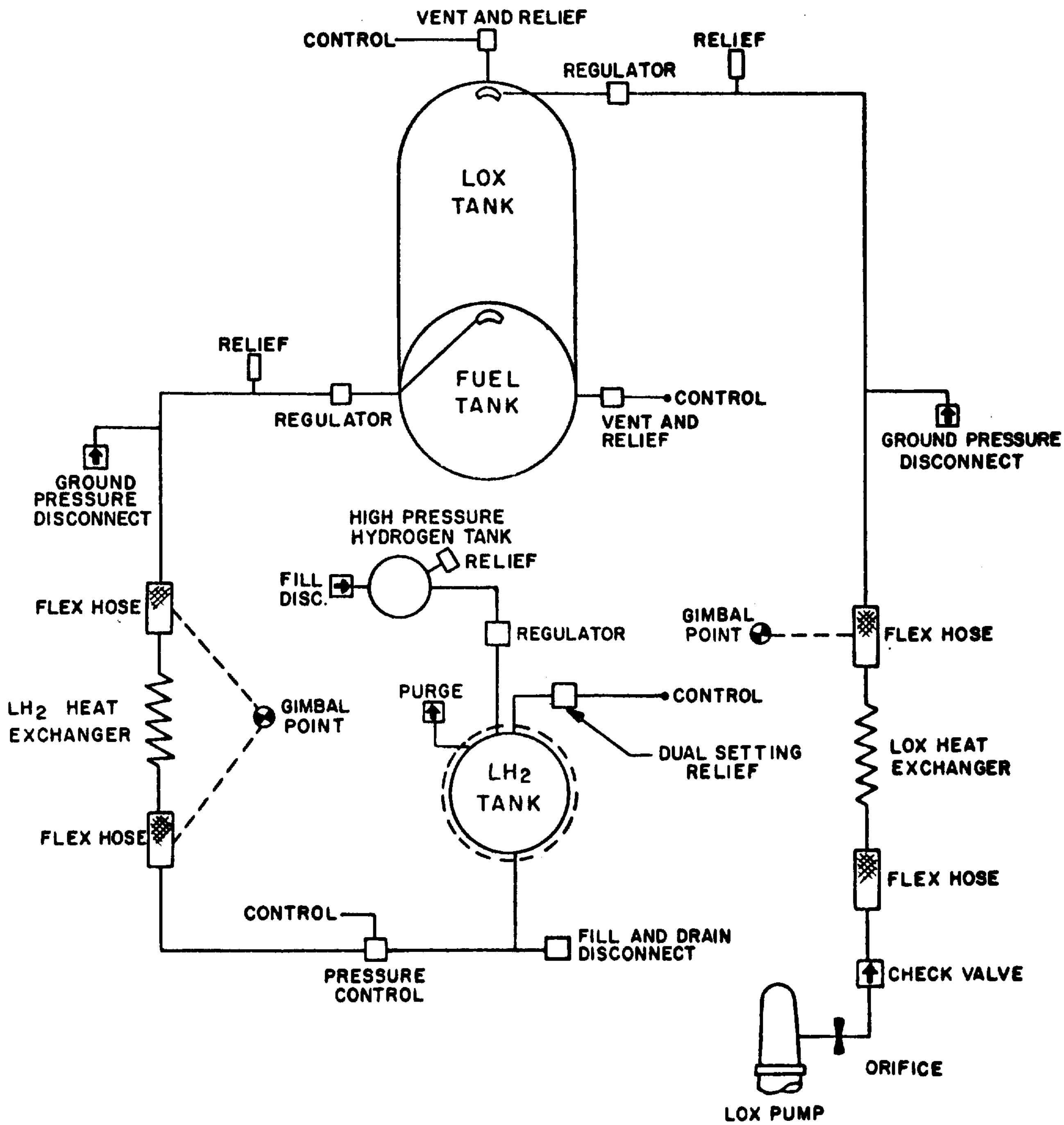


Figure 5. GOX LH_2 System, System 7

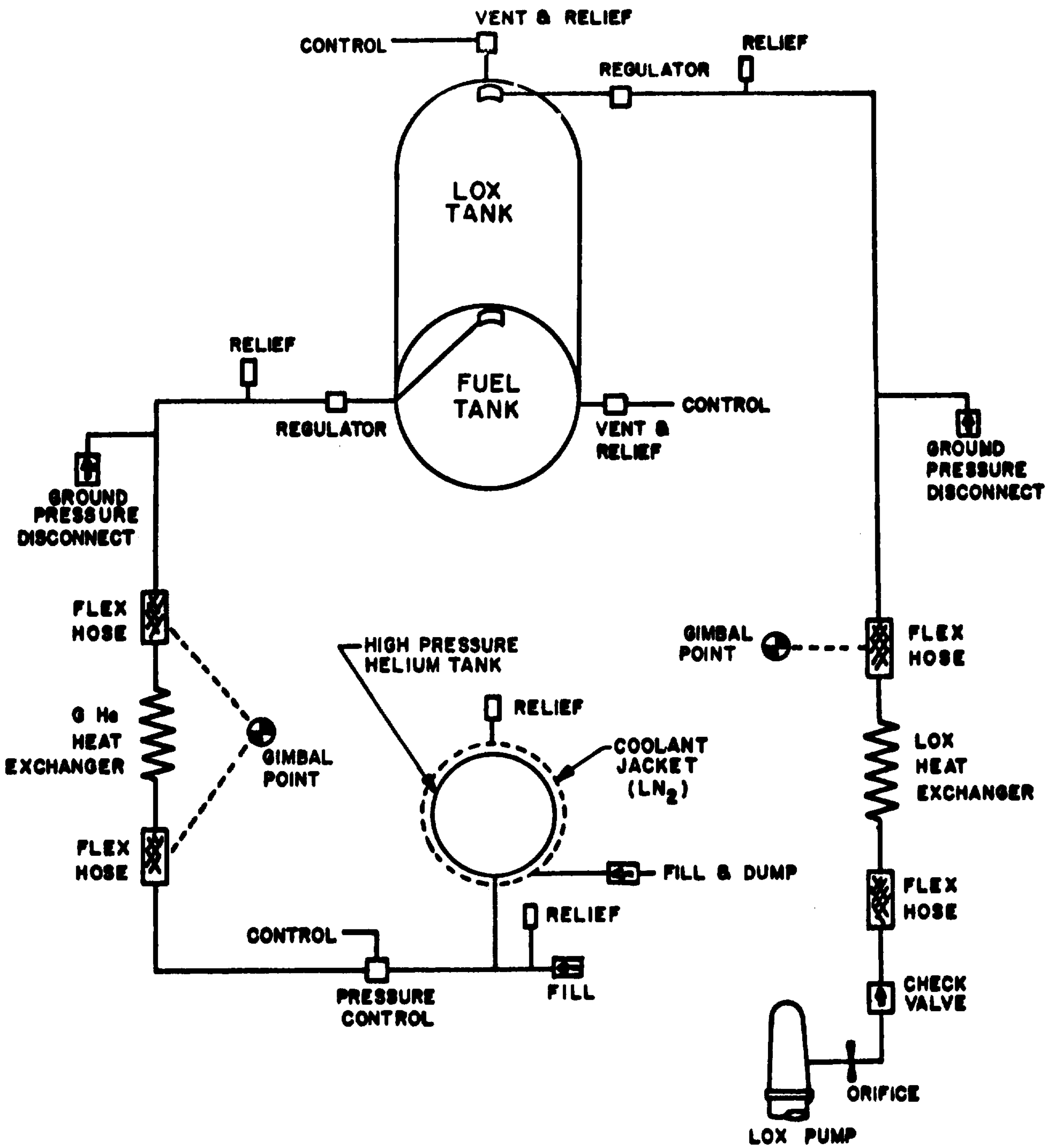


Figure 6. GOX/GHe System, System 8

For these systems, special provisions will have to be made to raise the storage tank pressure to its operating value prior to engine start. With these systems, and also with Systems 2, 6, and 7, there exists the posibility that additional liquified pressurant must be supplied if the missile countdown time exceeds a certain maximum.

In the oxidizer side of Systems 6, 7, and 8, LOX is bled off the engine LOX pump discharge and subsequently fed into the main heat exchanger.

In System 2 and the fuel side of System 6, nitrogen is loaded into a supply tank in the liquid state and at ambient pressure. A small nitrogen tank and pressure regulator are used to apply and maintain pressure in the supply tank for transfer of pressurant into the propellant tanks. Alternative methods considered included the use of an internal heat exchanger (as in Systems 1 and 3), and "self pressurization" of the supply tank by attaining equilibrium between the liquid and vapor phases causing evaporation of liquid nitrogen as the removal of liquid tends to reduce the pressure. These methods were found to be higher in weight and lower in reliability, and were not further considered.

Similarly, the liquid hydrogen of System 7 is expelled by a supply of high-pressure gaseous hydrogen stored at ambient temperature.

In general, the schematics show arrangements that conform to past practice. It was beyond the scope of this study to determine the optimal arrangement for the control valves.

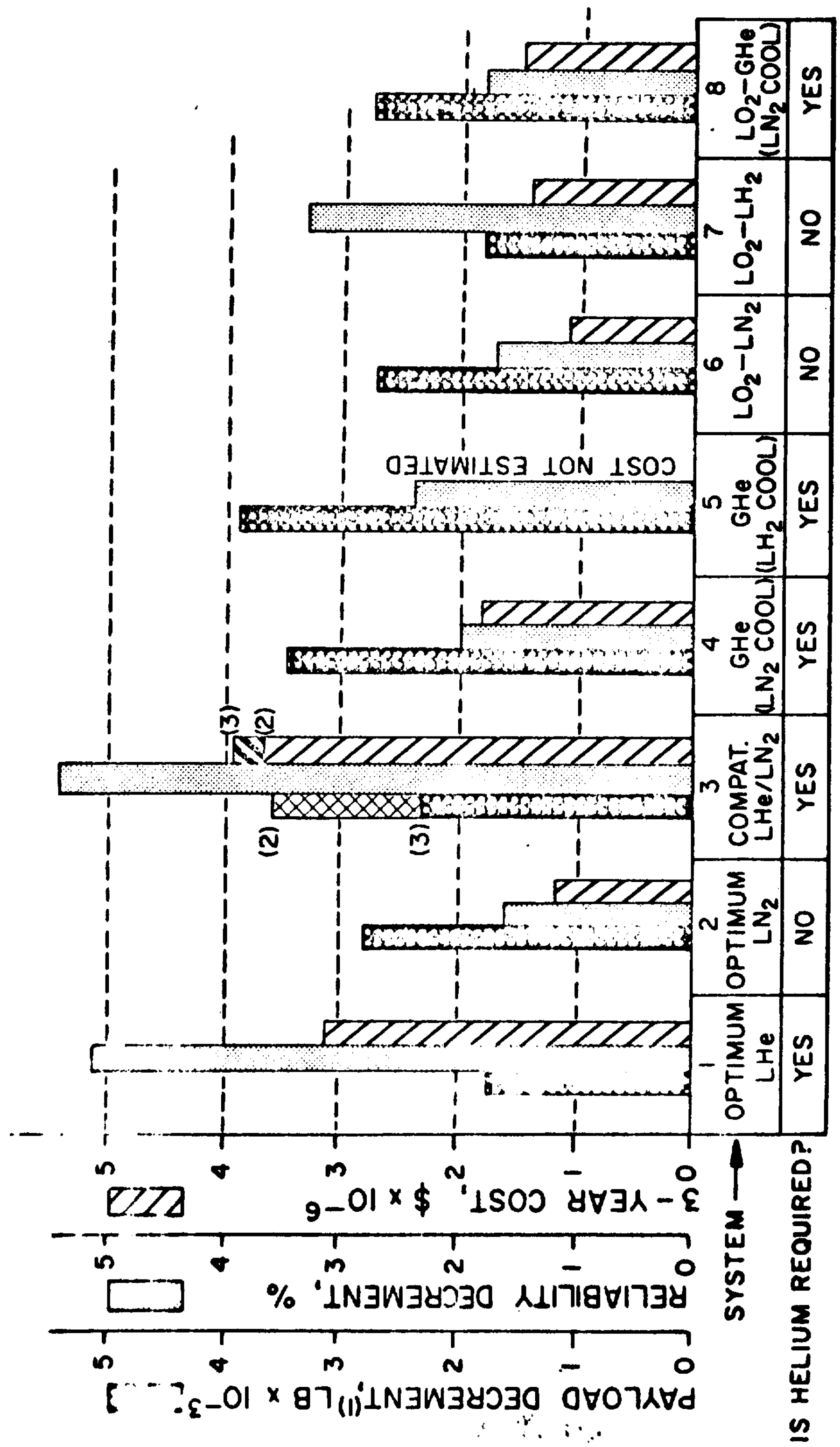
The main heat exchanger was assumed to be located in the turbine exhaust duct which is on the gimbaled portion of the engine. The schematics therefore show flex lines where connections must be made with the main heat exchanger across the gimbal plane.

To maintain storage tank pressure for the liquid hydrogen system (No. 7), only two gases could be considered, namely hydrogen and helium. The use of helium is questionable since only a few degrees rise in hydrogen bulk temperature could cause its density to fall below that of the helium near the interface. It was thus decided to utilize hydrogen gas at ambient temperature to expel the liquid hydrogen. This method has been successfully utilized to transfer liquid hydrogen during the ROVER program.

DETAILED RESULTS

Figure 7 presents an over-all comparison of the systems, while Table 1 shows a more detailed cost breakdown.

It was assumed that the pressurization system is intended to serve a single F-1 engine. In all probability, F-1 engines will also be used in clusters. Although the number of engines per vehicle may affect the magnitude of the figures, it is not expected to affect the relative ratings of the systems.



(1) Based on LOX/LH_2 second stage (see page 27)
(2) When using LN_2
(3) When using LHe

Figure 7. Over-All System Comparison

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TABLE 1

COST COMPARISON SUMMARY

(1000's of dollars)

SYSTEM	1 OPT LHe	2 OPT LN_2	3 COMPAT	4 GHe (LN_2 COOL)	5 GHe (LH_2 COOL)	6 LO_2-LN_2	7 LO_2-LH_2	8 LO_2-GHe
Added Development Costs:					NOT ESTIMATED	*		
Additional Facilities	414	0	414	122		0	116	122
Pressurant	989	0	1480	198		4	16	132
Total	1403	0	1894	320		4	132	254
Operational Costs:								
GSE, per set	392.5	139.5	465.0	295.0		119.5	145.0	215.0
Hardware, per set	87.3	78.5	99.3	87.3		73.3	87.0	77.3
Pressurant, per run	5.5	0.4	7.6* 0.2**	1.3		0.4	0.3	1.0
3-Year Total	3106	1173	3893* 3656**	1792		1089	1418	1506

* With LHe
** With LN_2

Although all systems studied were assumed to use propellant tank pressure regulators, some of the systems would allow use of orifices instead. The resultant weight penalties and reliability increases are summarized in Table 2.

Weight

Pressurant gas requirements were estimated by the methods outlined in Appendix C. The results of this analysis revealed that the optimum pressurizing gas delivery temperature is not necessarily the same for the LOX and fuel tanks. (Optimum in this report is used with regard to system weight only.) For the LOX tank it was found that the higher the delivery temperature, the lower the over-all system weight. A figure of 960 R (500 F) was chosen as optimum since this is felt to be the highest temperature compatible with tank materials and regulator and valve seals at the present time.

On the other hand, calculations show that in the fuel tanks, minimum pressurant gas and evaporated propellant weight occur at a gas delivery temperature of 560 R (100 F) for all systems except that utilizing nitrogen in the fuel tank, in which case the optimum temperature was 700 R (240 F). This results from the condition that, at higher temperatures, the weight of evaporated propellant nullifies the lower pressurant gas requirement. This does not mean, however, that the over-all system weight is lower, since the increased pressurant requirement for low delivery temperatures is also reflected in hardware weight. In view of this, it was decided to estimate system weights for each of the cases as follows:

a. 960 R gas to both tanks
b. Optimum to each tank
c. 560 R gas to both tanks

TABLE 2

SYSTEM COMPARISON, ORIFICES vs REGULATORS

	System Number							
	No. 2 - OPT LN_2		No. 6 - GOX/LN_2		No. 7 - GOX/LH_2		No. 8 - GOX/GHe*	
	Regulator	Orifice	Regulator	Orifice	Regulator	Orifice	Regulator	Orifice
Weight - lbs	3125	3300	3017	3283	2006	2321	3036	3351
Payload Weight Decrement - lbs**	2812	2970	2716	2940	1805	2090	2732	3010
Reliability Decrement, percent	1.6	1.3	1.7	1.4	3.3	3.1	1.8	1.6
Hardware Cost (Thousands of Dollars) per system	79	63	73	57	87	75	77	69
Pressurant Cost (Dollars) per firing	400	500	350	450	250	300	950	1000

* Orifice on GOX side only
**Based on a $LOX-LH_2$ second stage

NOTE: Weight disadvantage using orifices is not as great as indicated since the additional pressurant is vented overboard and cannot be considered "dead weight" during the entire flight. This would more appropriately be stated in terms of an I_{sp} loss, but these calculations were not made for this report.

In every case it was found that there is only a negligible over-all weight difference between (a) and (b), so for simplicity in system design a gas delivery temperature of 960 R was chosen for both tanks for those systems utilizing the same pressurant supply for both tanks.

Case (c) was considered since it simplifies heat exchanger, regulator, and seal design. However, the system weight increase outweighed simplification.

For "dual" pressurization systems (i.e., different fluids to each tank) the "optimum" delivery temperature was used to each tank. In this case the slight over-all weight advantage obtained by utilizing high delivery temperatures to the fuel tank is more than balanced by the advantage of low temperatures in heat exchanger, regulator, and seal design. A summary graph showing over-all system weights is given in Fig. 8.

On the basis of the perfect gas law, it would seem that System 5 should be superior to System 4, since, when helium gas is stored at liquid-hydrogen temperature (36 R), it should require a considerably smaller storage tank than when stored at liquid nitrogen temperature (140 R), and that this should be reflected in an over-all weight savings for the same storage pressure. This is not, however, the case, since helium compressibility factors in this temperature and pressure region are extremely high [compressibility factor is defined by $Z = (PV)/(WRT)$]. For a 3250 psi storage pressure, the compressibility factor is about 1.36 at 140 R, but at 36 R the value is 2.51. As it turns out, the storage tank is actually the same size (and thus weight) in either case, and the amount of residual helium using hydrogen cooling is 558 lb compared with 162 lb using nitrogen cooling.

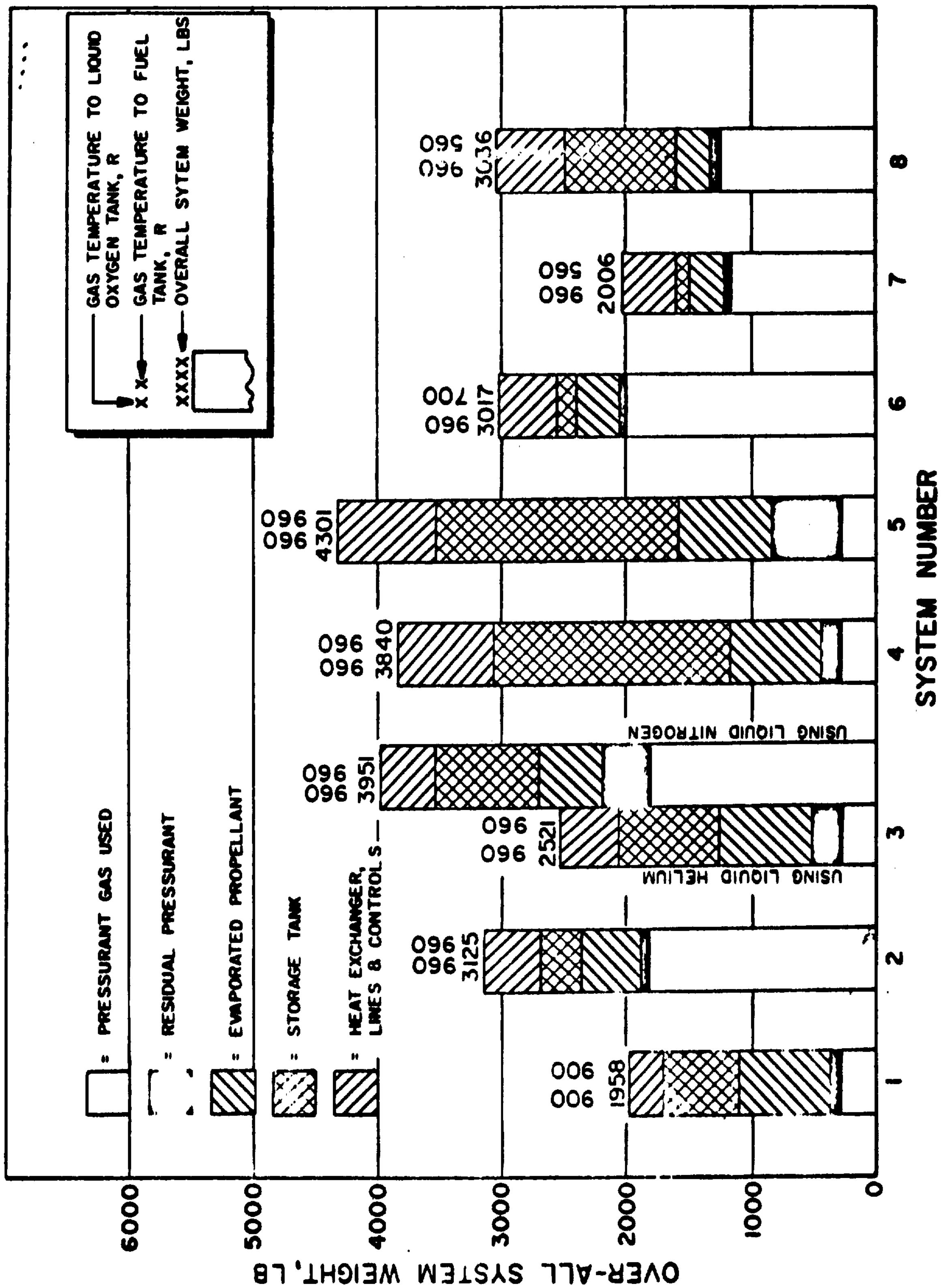


Figure 8. System Weight Comparison

This led to an analysis of the optimum storage temperature for a high-pressure helium system. The results are summarized in Fig. 9. The optimum temperature is very close to the normal boiling point of liquid nitrogen.

Secondly, it is seen that System 4 has one of the highest over-all weights. However, this system was picked for the Atlas vehicle, since the Atlas carries the booster main propellant tanks through the sustainer phase, the booster engines and pressurant storage system being jettisoned at staging. Range exchange figures for the Atlas show that 1 lb of dead weight in the sustainer phase causes a range decrease of 0.36 n.mi while 1 lb of jettisonable weight results in a range decrease of only 0.09 n.mi.

Of the items shown in Fig. 8, only the pressurant in the main tanks and the evaporated propellant are present in the vehicle after staging; the storage tanks, heat exchanger, residual helium, and most of the controls may be jettisoned. Figure 10 shows a comparison of the pressurizing systems on this basis.

The "effective" weight is defined as that which could be carried as jettisonable dead weight during boost phase with the same effect on range as the system being considered. Since neither liquid hydrogen nor helium was available in large quantities at the inception of the Atlas program, it is doubtful that Systems 1, 3, and 7 were even considered.

From this discussion it is seen from Fig. 10 that on the basis of jettisonable weight, the choice of the Atlas system was a sound decision at the time it was made.

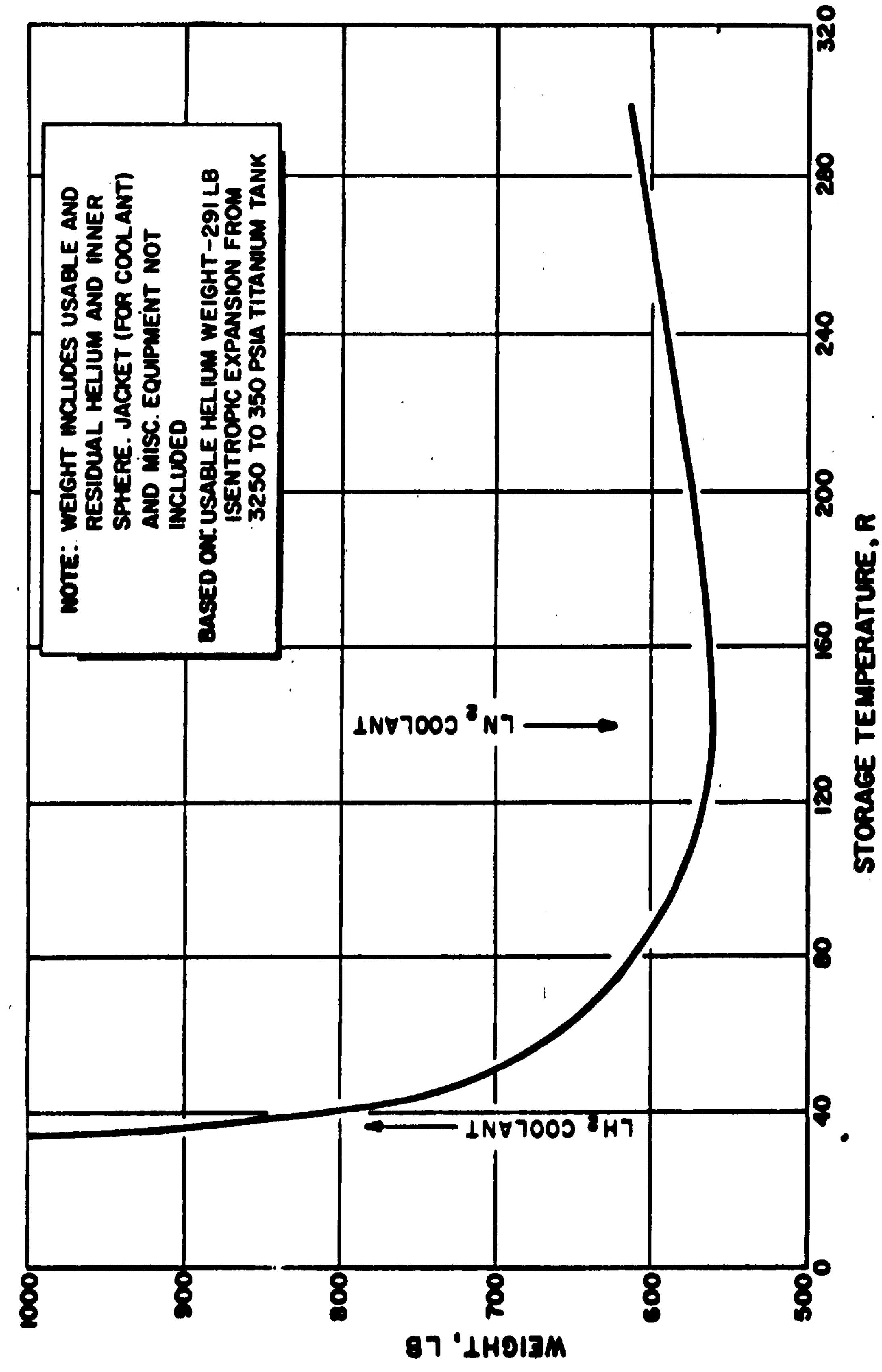


Figure 9. Weight Comparison of High Pressure Gaseous Helium System as a Function of Storage Temperature

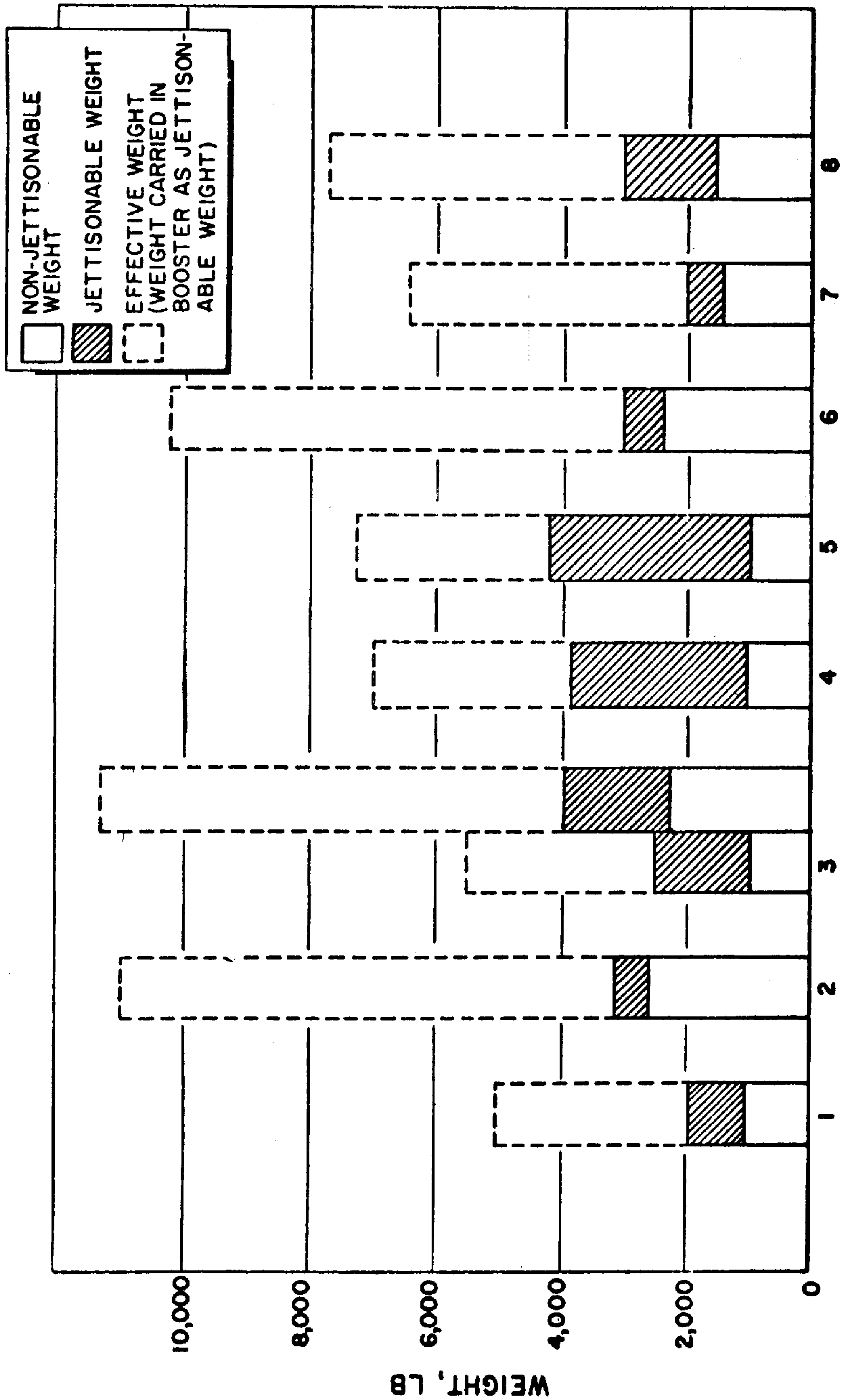


Figure 10. F-1 System Weight Comparison if Used on Atlas-Type Vehicle

It is assumed, however, that the vehicle to be powered by F-1 engines is not a stage-and-a-half vehicle like the Atlas. Payload exchange factors (Ref. A8) for an F-1 powered vehicle are:

Mission	-	300 n.mi orbit
Vehicle	-	4,000,000 lb gross weight
First Stage	-	4 F-1 engines (LOX/RP-1)
Second Stage	-	One LOX/RP-1 engine or one LOX/LH_2 engine with 1,600,000 lb vacuum thrust

	LOX/RP-1 Second Stage	LOX/LH_2 Second Stage
Second-Stage Gross Payload Weight, lb	125,000	240,000
Payload Decrement for 1000-lb Increase in Each First-Stage Engine Weight	560 lb = 0.45%	900 lb = 0.37%

The payload decrement for each system was calculated and is summarised in Fig. 7. The heaviest system (No. 5) results in a payload decrement of 2111 lb over the lightest system, based on a LOX/LH_2 second stage.

Costs

Estimated costs are shown in Table 1. The estimates were made by the method presented in Appendix G, where a more detailed cost breakdown is given. Hardware cost is the estimated basic manufacturing cost of a set of system components (including component functional tests) for small quantity production. System assembly cost is not included. Pressurant cost is based on a single firing (including losses due to evaporation of the liquified gas pressurants during the filling and chill-down of pressurant tanks). Three-year total cost includes the additional development cost and the cost of operating the system during the first three years of missile operation (Fig. 20).

Reliability

For the purposes of this study, reliability is defined as the probability of the system performing within model specification requirements during an actual launching. With a product improvement program, reliability would increase as a function of time. However, the figures shown are for the end of the previously specified development period, i.e., through PFRT. Details of the reliability estimates are discussed in Appendix F.

State of the Art

State of the art was estimated as discussed in Appendix E. The ratings shown in Table 4, corresponding to excellent (E), good (G), fair (F), and poor (P), reflect the estimate of current conditions in the industry. The ratings are strongly reflected in the reliability and cost estimates of Fig. 7.

State-of-the-art techniques associated with the various systems are relatively good except for the handling of liquid helium and the design and development of high-temperature, high-capacity gas flow controls for hydrogen. These deficiencies are due to the lack of past experience in these fields and could be overcome in time.

As a matter of interest, the pressurization methods used for current large liquid propellant rocket engines is included in Table 3.

TABLE 3

PRESSURIZATION SYSTEMS USED IN CURRENT MISSILES

Missile	Oxidizer Tank			Fuel Tank		
	Pressurant	Inlet Temperature, F	Control	Pressurant	Inlet Temperature, F	Control
Redstone	Vaporized LOX	About -100	Orifice and Relief Valve	High-Pressure Nitrogen	About 200	Pressure Switch and Solenoid Valve
Jupiter	Vaporized LOX	About -100	Orifice and Relief Valve	High-Pressure Nitrogen	Ambient	Pressure Switch and Solenoid Valve
Thor	Vaporized LOX	About -100	Orifice and Relief Valve	High-Pressure Nitrogen		Orifice
Atlas Booster	High-Pressure Helium	300 to 550	Regulator	High-Pressure Helium	300 to 550	Regulator
Atlas Sustainer	None	-	-	None	-	-
Titan I Stage One	High-Pressure Helium	Above Ambient	Regulator	High-Pressure Helium	Above Ambient	Regulator
Titan I Stage Two	High-Pressure Helium	Above Ambient	Regulator	High-Pressure Helium	Above Ambient	Regulator
Saturn Booster	Vaporized LOX	About 80	Orifices and Relief Valves	High-Pressure Nitrogen	Ambient	Pressure Switches and Solenoid Valves

APPENDIXES

A. References

B. Ground Rules

C. Pressurant Requirements

D. Weight Estimates

E. State of the Art

F. Reliability

G. Cost Estimates

H. Liquid Helium Storage and Handling

APPENDIX A

REFERENCES

A. GENERAL

1. R-1559, "F-1 Tank Pressurization Study," Rocketdyne, A Division of North American Aviation, Inc., 15 May 1959.

2. R-1171P, "Program for Development of a 1,000,000 lbs Thrust Liquid Propellant Rocket Engine," Rocketdyne, A Division of North American, Inc., 10 November 1958.

3. McAdams, W. H., "Heat Transmission," 3rd ed., McGraw Hill, New York, 1954.

4. "Chemical Engineers Handbook," 3rd ed., McGraw Hill, New York 1950.

5. Roark, R. J., "Formulas for Stress and Strain," 3rd ed., McGraw Hills, New York, 1954.

6. Jenkins, D., "General Requirements for a Hydrocarbon Fuel-Liquid Oxygen Rocket Engine System" Convair Astronautics Report ZK-7-011, 1955.

7. ADM 60-74, V. Larson to P. Dietz, "Mission Profile for F-1 Engine," 10 May 1960.

8. AD-60-308-A, "Booster Exchange Factors" H. J. Wieseneck, 27 May 1960 (revision)

B. FLUID PROPERTIES

1. Chelton and Mann, "Cryogenic Data Book," University of California Radiation Laboratory, Report 3421, May 1956.

2. Simmons, J. T., "The Physical and Thermodynamic Properties of Helium," Whittaker Co., Ltd., Report D-9027, 1 July 1957.

3. "Propellant Properties Manual," Research Report 59-48, Rocketdyne, A Division of North American Aviation, Inc., 1 Feb. 1960.

C. PROPELLANT HANDLING AND STORAGE

1. Scott, L. E., Cryogenic Engineering Laboratory, National Bureau of Standards, Boulder Colorado, Project 81426 "Evaluation of Three Technical Proposals for F-1 Pressurization System," 6 May 1960.

2. "Proceedings of the 1958 Cryogenic Engineering Conference," 3-5 September 1958.

3. "Cryogenic Piping System Design Considerations," Jacobs, R. B., Heating, Piping and Air Conditioning, February, 1960.

4. "Low Temperature Technology - What the Latest?", Chemical Engineering, 30 November 1959.

D. PRESSURANT REQUIREMENTS

1. Tuttobene, R. and D. L. Greene, "Intermediate Stub Tank Pressurization Tests," Convair Astronautics Report No. 9412-5, 10 May 1957.

2. LEM-848-1101, Nomad Development Unit, "Hot Helium Pressurizing Tests," 17 October 1958.

3. Yip, P. S., and M. T. Norton, "A Study of the Helium Weight Requirements, Temperature and Pressure Histories of the LOX and JP-4 Tanks During Powered Flight of the XSM-65B Missile," Convair Astronautics Report ZJ-7-015, 12 August 1955.

4. Yip, P. S. "A Study of the XSM-65B LOX and Fuel Tank Skin Temperatures Due to Aerodynamic Heating," Convair Astronautics Report ZJ-7-012, 1955.

5. Wilson, C. E., "Temperature Rise of the Liquid Oxygen and Chilling of the Pressurizing Helium for Insulated and Uninsulated Intermediate Bulkheads of the SM-650," Convair Astronautics Report ZJ-7-050, 3 December 1957.

6. Wilson, G. C., "Operational Missile Liquid Oxygen and Fuel Tank Skin Temperatures Due to Aerodynamic Heating and Helium Weight Requirements During Powered Flight," Convair Astronautics Report ZJ-7-047, 16 October 1957.

7. "Project Atlas Summary Report," Vol. III, Thermodynamics, Convair Astronautics Report ZM-7-011, 1 January 1952.

APPENDIX B

GROUND RULES

The present tank pressurization study was based on the following ground rules:

1. The vehicle is a first-stage booster powered by a single F-1 engine.
2. The engine thrust is 1,500,000 lb at sea level.
3. The propellants are LOX and RP-1.
4. The nominal propellant flowrates are 3920 lb/sec of LOX and 1742 lb/sec of fuel, consistent with present F-1 engine design.
5. The duration is 150 sec.
6. The propellant densities at lift-off are 50.45 lb/ft^3 for fuel (the mean density at 60 F) and 70.50 lb/ft^3 for LOX. The latter value is an estimate based on the average of 70.92 lb/ft^3 experienced at lift-off of Thor flights and the average of 70.16 lb/ft^3 experienced at lift-off of Atlas D Series flights.
7. The tank configuration is shown in Fig. 11. The fuel tank shown includes an allowance of 7.6 percent, and the oxidizer tank an allowance of 4.8 percent for reserve propellants and ullage.
8. The required minimum NPSH is 71 ft for the LOX pump, 114 ft for the fuel pump, in agreement with the Model Specifications.
9. Boiling in the LOX tank will not necessarily be suppressed, except as necessary to comply with the minimum required NPSH.

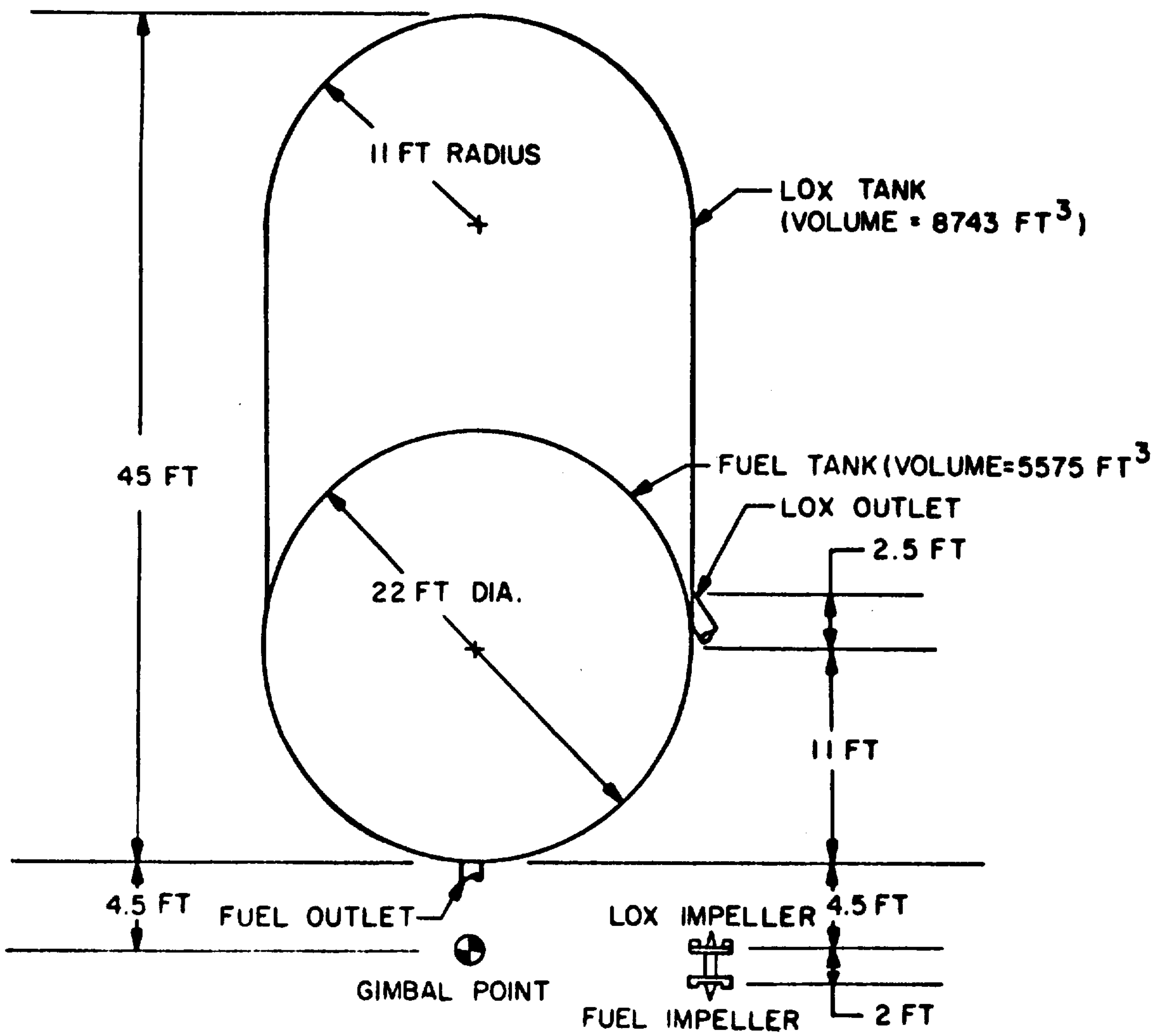


Figure 11. Assumed Tank Configuration and Assumed Dimensions for Single F-1 Engine Vehicle

10. Suction line losses at the nominal flowrates and the densities given above are 15 psi in the LOX line and 10 psi in the fuel line, between the respective tank and pump impeller inlet. These values are based on the position of the engine gimbal point shown in Fig. 11, a single 17-in. diameter LOX duct, two 12-in. diameter fuel ducts and estimated line lengths and configurations.

11. The missile total acceleration at engine cutoff is 5.4g (Ref. A7).

12. The tank pressures determined on the basis of the pump NPSH requirements are adequate for structural purposes.

13. Lube tank pressurization, seal purge and possible other minor uses of pressurant gas are negligible.

14. The tank ullage space is pressurized by GSE prior to lift-off.

The fuel and LOX tanks pressure histories used throughout the present studies are given in Fig. 12. The acceleration curve shown in this figure was taken from Ref. A7. This curve was used to compute the required tank pressure histories, as follows:

$$P_T = P_V + \Delta P_L + \frac{\rho H_{NS}}{144} - \frac{\rho a h}{144}$$

where

P_T = tank pressure, psia

P_V = propellant vapor pressure, psia

ΔP_L = suction line pressure drop, psi

ρ = propellant density, lb/ft^3

H_{NS} = required NPSH, ft

a = vehicle total acceleration, g's

h = height from pump inlet to liquid surface, ft

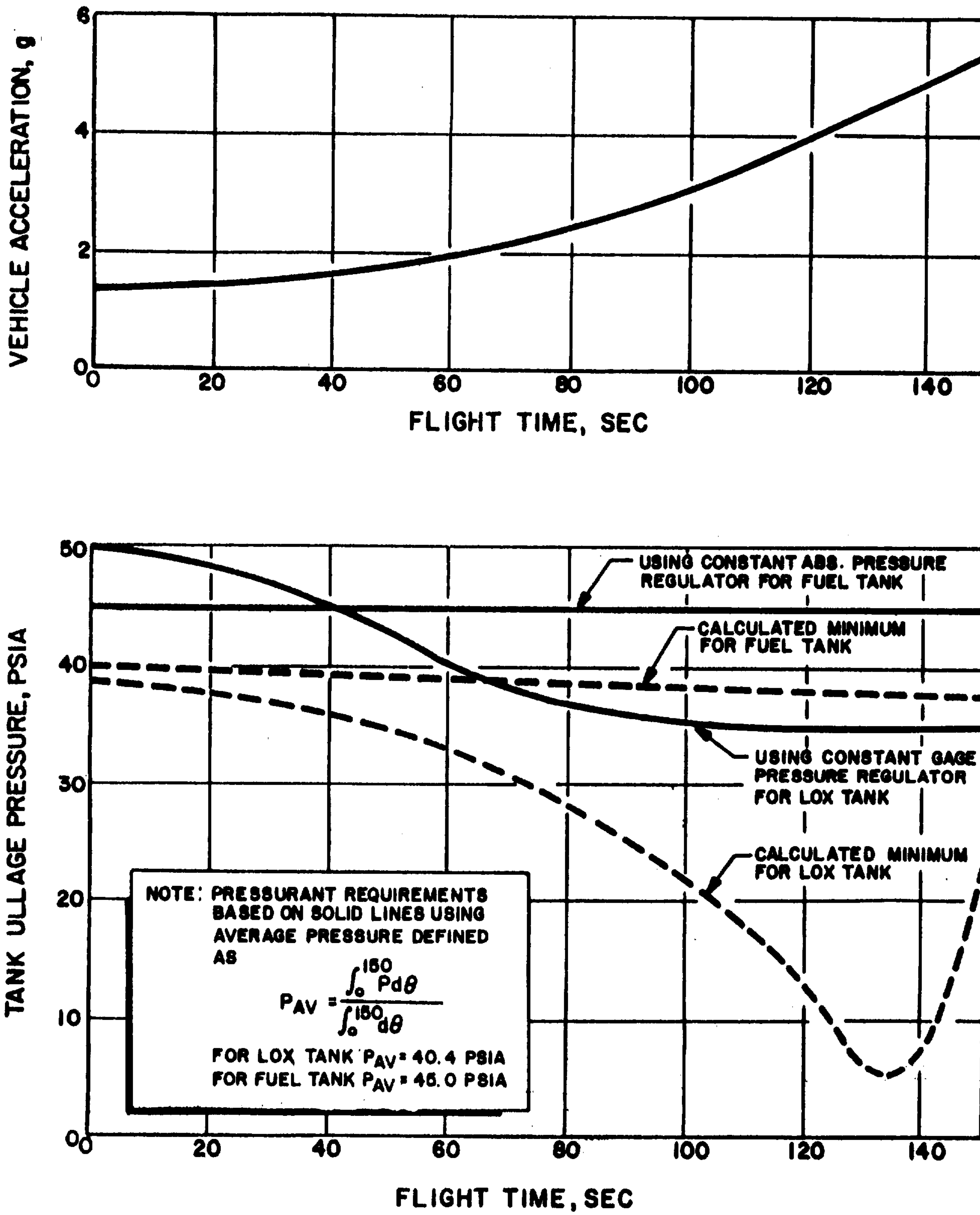


Figure 12. Assumed Acceleration Curve and Propellant Tank Pressure Histories

The height, h, of the propellant surface in the tank above the pump inlet was computed on the assumption of constant volumetric propellant flowrates.

The required tank pressure histories computed in this manner are shown in Fig. 12. The required fuel tank pressure is essentially constant. Hence, an absolute-pressure regulator was assumed for the fuel tank. The required LOX tank pressure at burnout is appreciably below that at lift-off. A gage pressure regulator was therefore assumed for the LOX tank, it being the simplest control having approximately the desired characteristic. The settings of the regulators were assumed such as to provide a minimum margin of 5 psi between the required and the actual pressure in each tank, as indicated in Fig. 12.

It is evident that the actual amount of stored pressurant is dependent only on the final conditions prevailing in the main propellant tank. The determination of these conditions, however, requires a knowledge of the temperature and flowrate histories in the tank throughout the flight. Accurate estimation of these parameters is possible only by a numerical integration process which was deemed to be outside the scope of this study. It was thus decided to base the main LOX tank pressurant requirements on an integrated average tank pressure as defined in Fig. 12. Results obtained in this manner are felt to be conservative.

APPENDIX C

PRESSURANT REQUIREMENTS

LOX TANK

The estimation of pressurizing gas requirements and the effect of heat exchanger outlet temperature on this quantity was found to be an extremely complex problem.

A literature survey was made and, although considerable data are available, none present a suitable mathematical model for predicting the experimental results. It was also found that in most of the references at least one critical parameter was either not measured or the measurement was lost due to instrumentation failure.

Several qualitative conclusions, however, may be reached from these reports as follows:

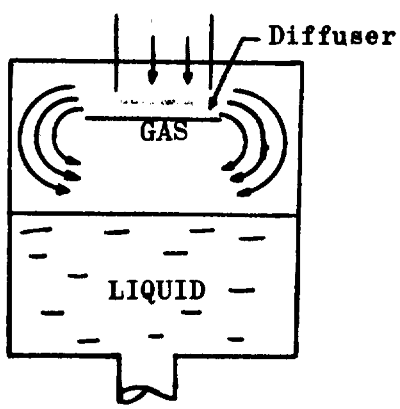
1. There is severe temperature stratification in the main tank ullage space. Further, the temperature distribution may be time dependent.
2. The installation of a pressurizing gas diffuser at the inlet to the tank is necessary and its design is critical.
3. There is simultaneous heat and mass transfer between the pressurizing gas and bulk liquid.

4. Aerodynamic heating of both the pressurizing gas and bulk liquid greatly affects the gas requirements on the Atlas vehicle. Although its effect on the final F-1 vehicle will probably be smaller, it may not be negligible.

5. The initial ullage volume and hold time affect the amount of gas required if the tanks are pre-pressurized from a ground supply, (i.e., a large initial ullage volume in the LOX tank reduces flight pressurizing gas requirements).

6. Intertank heat transfer may be an important factor.

In view of the above, it was decided to construct a mathematical model of the tank pressurizing phenomena based on an assumed mechanism as described below.



In establishing the equations, it was assumed that the pressurizing gas requirements could be based on an average temperature in the ullage space. The rate at which heat is transferred to the bulk fluid could not be estimated by natural convection coefficients or by molecular diffusion theory.

It was thus concluded that the installation of a diffuser probably caused the pressurizing gas to circulate as shown in the diagram, resulting in turbulent heat and mass transfer coefficients at the gas-to-liquid interface. Since the transfer rates are dependent on the characteristics of the boundary layer, and these characteristics can not be determined from available data, it was decided to formulate the process and solve for a heat transfer coefficient from the equation:

$$h = \frac{q}{A\left(T_u - T_s\right)} \qquad (1)$$

where

q = heat transfer rate necessary to evaporate the amount of propellant estimated from the data, Btu/sec

A = tank cross sectional area, ft^2

T_u = average ullage temperature, R

T_s = fluid saturation temperature, R

h = heat transfer coefficient, Btu/sec ft^2 R

This equation assumes the following:

1. All of the heat transferred goes to evaporating propellant.
2. The heat transfer area is the cross sectional area of the tank (i.e., no evaporation takes place from the tank walls or interfacial disturbances).
3. That the driving force is between the "average" ullage temperature and the bulk fluid saturation temperature.
4. Tank dimensions (except cross sectional area) have no effect on heat transfer.
5. There is no heat transfer through the tank walls, including absence of aerodynamic heating.

While the above assumptions do not constitute a rigorous description of the actual mechanism, they represent the best currently available approach.

A brief discussion of the most useful references and the heat transfer coefficients obtained from them is given below. A complete list of references used is included in Appendix A.

Convair Stub Tank Tests (Ref. D1)

Liquid nitrogen was discharged from a scaled-down Atlas tank by the use of helium at ambient temperature. Temperature stratification was monitored by means of thermocouple rakes placed in the tank at various locations. Relative amounts of vaporized nitrogen and helium were monitored by means of gas sampling probes mounted at various points along the tank wall. Actual pressurant gas flow measurement was attempted by use of an orifice, but this information was lost due to instrument failure.

Based on these data, the heat transfer coefficient was estimated to be 0.0036 Btu/sec ft^2 R.

Nomad Tank Pressurization Tests (Ref. D2)

Liquid nitrogen was expelled from a spherical tank by helium gas at about 200 F. A thermocouple rake was used to monitor temperature stratification in the tank but no measurement of actual gas or evaporated propellant quantities was attempted.

Relative weights were estimated assuming pure helium in the ullage volume and the temperature distribution measured therein. A plot of temperature distribution in the ullage volume at the end of the test run is shown in Fig. 13.

Based on these figures, a heat transfer coefficient of about 0.005 Btu/sec ft^2 R was obtained.

<u>ABMA Data</u>

A curve of average ullage temperature vs inlet gas temperature for a GOX-on-LOX system was obtained from ABMA (now MSFC) during the course of the study. This plot is reproduced in Fig. 14.

The information was obtained from flight test data over a range of gas temperatures of 300 to 450 R and extrapolated by ABMA to cover the range shown.

Since no measurements of pressurizing gas and evaporated propellant quantities were available, a series of curves was obtained for various values of h using the formulated equations. Results are shown by the dotted lines of Fig. 14. The general shape and magnitude of the theoretical curves as compared with the data is remarkable considering the assumptions. Since the figure of h = 0.002 fits the data over the experimental range it was felt that this value satisfactorily approximates the data.

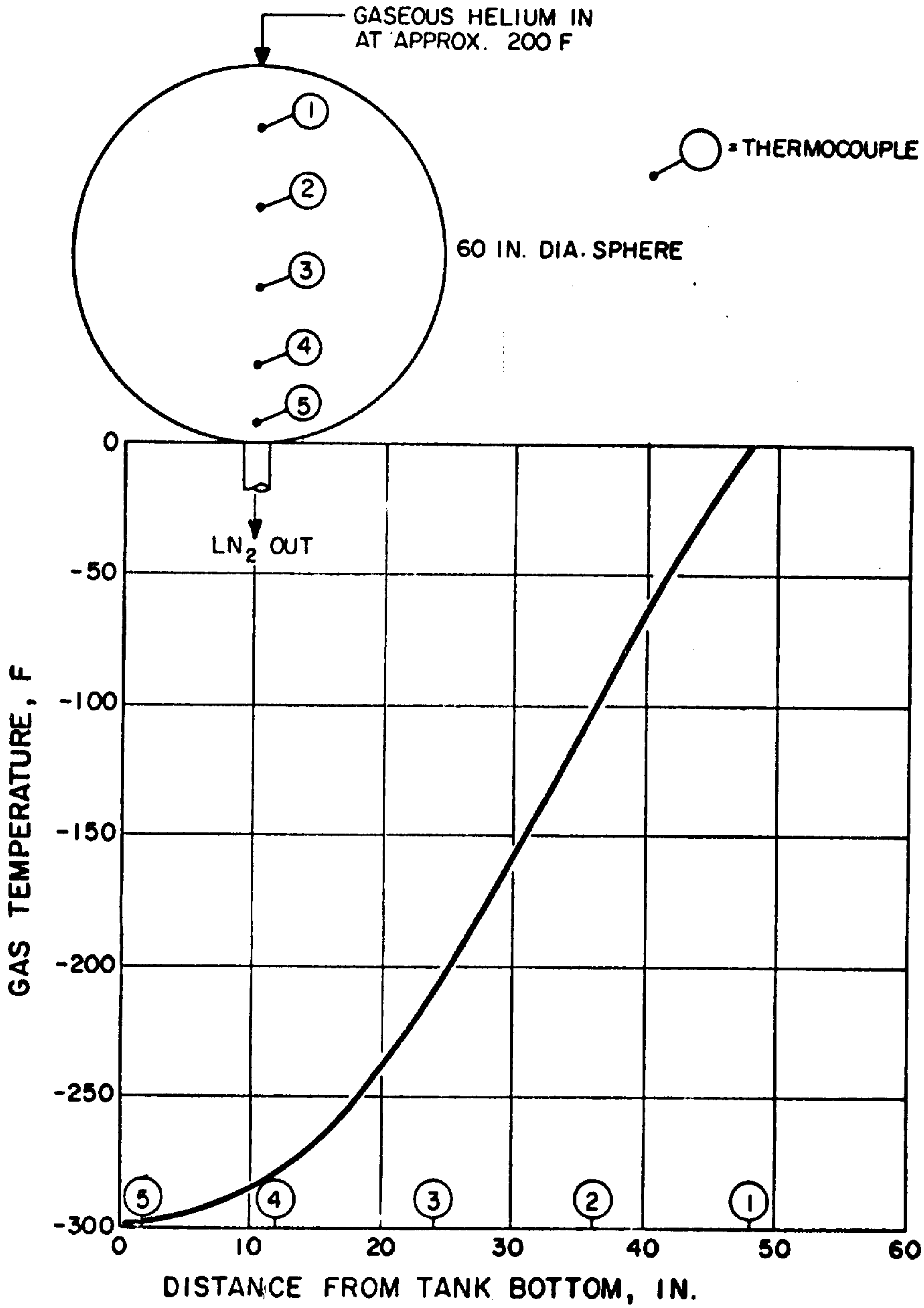


Figure 13. Final Temperature Distribution in Nomad Tank Expulsion Study

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[redacted]

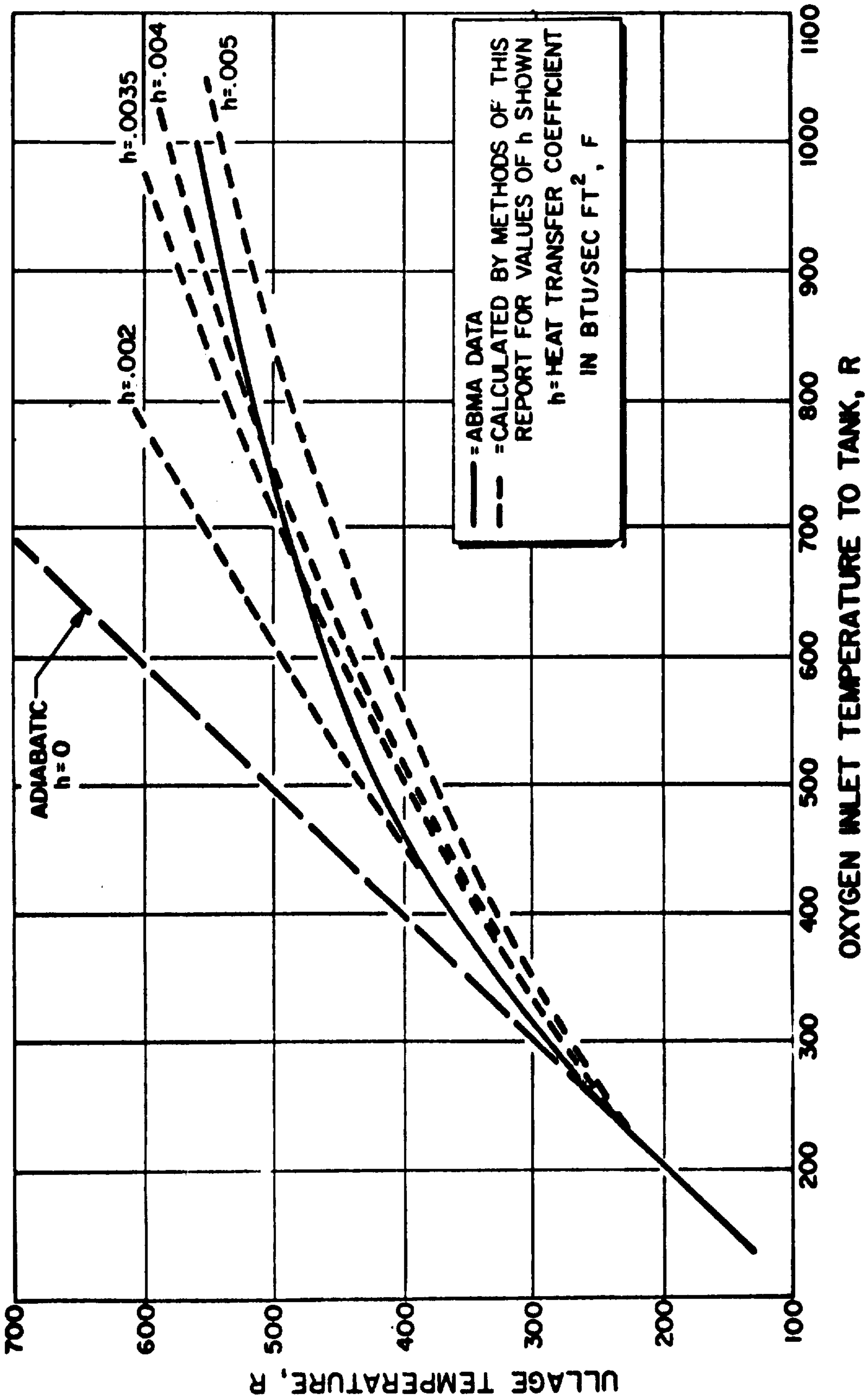


Figure 14. Comparison of Calculation Method With ABMA Data for Pressurization of Jupiter Main LOX Tank With GOX

[redacted]

From the above it was decided to use the following values of h in the F-1 study.

A. For helium and hydrogen on LOX, $h = 0.005$ Btu/sec ft^2 F

B. For nitrogen and oxygen on LOX, $h = 0.002$ Btu/sec ft^2 F

The calculations were made in the following order: (nomenclature, page 52)

1. Choose a value of T_u

2. $q = hA\,(T_u - T_S)$

3. $\dot{W}_{EV} = q \Big/ \left[\lambda_V + C_L\,(T_S - T_L) + C_{P_{EV}}\,(T_u - T_S) \right]$

4. $\dot{V}_{EV} = Z_{EV}\,\dot{W}_{EV}\,R_{EV}\,T_u / P_t$

5. $\dot{V}_G = \dot{V}_E - \dot{W}_{EV}/\rho_L - \dot{V}_{EV}$

6. $\dot{W}_G = P_t\,\dot{V}_G / Z_G\,R_G\,T_u$

7. $Y = \dot{W}_{EV}/\dot{W}_G$

8. $T_G = \left\{ Y\left[\lambda_V + C_L\,(T_S - T_L) + C_{P_{EV}}\,(T_u - T_S) \right] + C_{P_G}\,T_u \right\} / C_{P_G}$

In this manner curves of $\dot{W}_{EV}$, $\dot{W}_G$ and T_u were generated as functions of T_G and the optimum temperature determined.

FUEL TANK

The same problems exist in the fuel tank calculations as were encountered in those for the LOX tank with respect to determining the controlling mechanism. In the case of the fuel tank, since T_S is greater than T_u for the ranges of gas inlet temperature investigated, the mechanism and calculation procedure used for the LOX tank were not applicable.

Fuel tank calculations were therefore based on the assumption that the partial pressure of the fuel in the ullage volume is equal to its vapor pressure at the ullage temperature.

The calculation procedure was as follows:

1. Assume a value of T_u
2. Read P_{ev} at T_u from a vapor pressure plot for RP-1.
3. $P_g = P_t - P_{ev}$
4. $\dot{W}_{EV} = P_{EV} \dot{V}_E / R_{EV} T_u$
5. $\dot{W}_G = P_g \dot{V}_E / R_G T_u$
6. $T_G = \left\{ \dot{W}_{EV} \left[\lambda_V + C_{P_{EV}} (T_u - T_L) \right] + \dot{W}_G C_{P_G} T_u \right\} / \dot{W}_G C_{P_G}$

The resulting data were then treated as outlined for the LOX tank.

General

Results of the computations made by the methods described above are shown in Fig. 15, for system number 6, and are typical of results obtained for all pressurants studied.

As a matter of interest a graph showing the variation of required pressurizing gas and evaporated propellant as a function of the heat transfer coefficient is given in Fig. 16 for the main LOX tank pressurized with helium, nitrogen and oxygen at 960 R. This graph indicates that, if the assumed heat transfer mechanism is correct, the amount of pressurant gas required is a very weak function of h and should thus be predictable with fair accuracy. The same is not true of the evaporation rate and thus the final burnout weight may be considerably affected.

Nomenclature

A = tank cross sectional area, ft^2

C_L = specific heat of liquid, Btu/lb R

C_p = specific heat at constant pressure, Btu/lb R

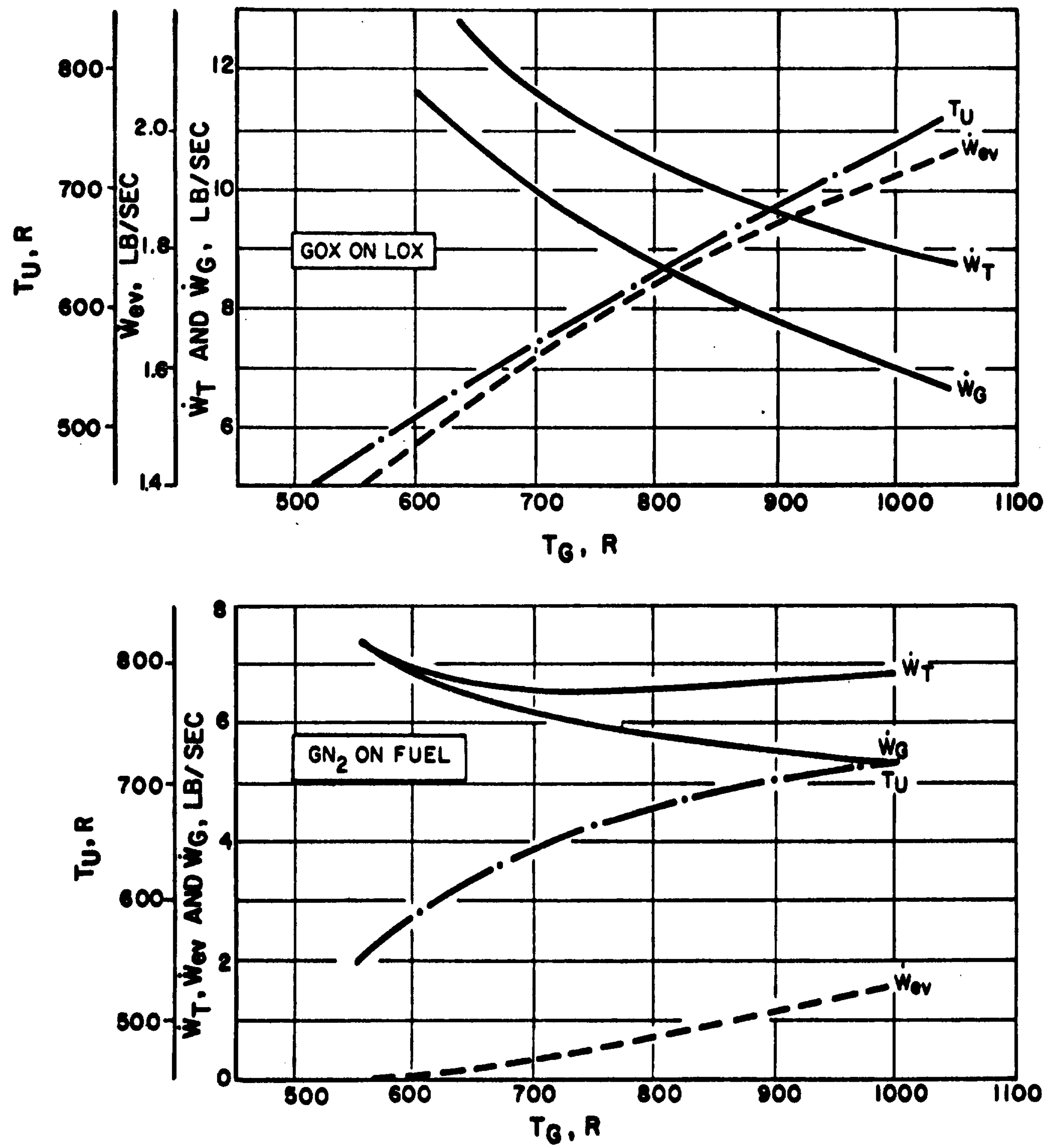
h = heat transfer coefficient, Btu/sec ft^2 R

P_t = main propellant tank pressure, lb/ft^2

P_{ev} = vapor pressure of fuel at T_u, lb/ft^2

P_g = partial pressure of pressurant gas defined by $P_g = P_t - P_{ev}$ for the fuel tank, lb/ft^2

q = heat transfer rate, Btu/sec



T_U = ULLAGE TEMPERATURE, R

T_G = PRESSURANT GAS INLET TEMPERATURE TO TANK, R

$\dot{W}_{ev}$ = EVAPORATION RATE, LB/SEC

W_G = PRESSURANT FLOWRATE LB/SEC

$W_T = W_{ev} + \dot{W}_G$ = TOTAL FLOWRATE LB/SEC

Figure 15. Results of Pressurant Calculations for GOX on LOX - N_2 on Fuel System

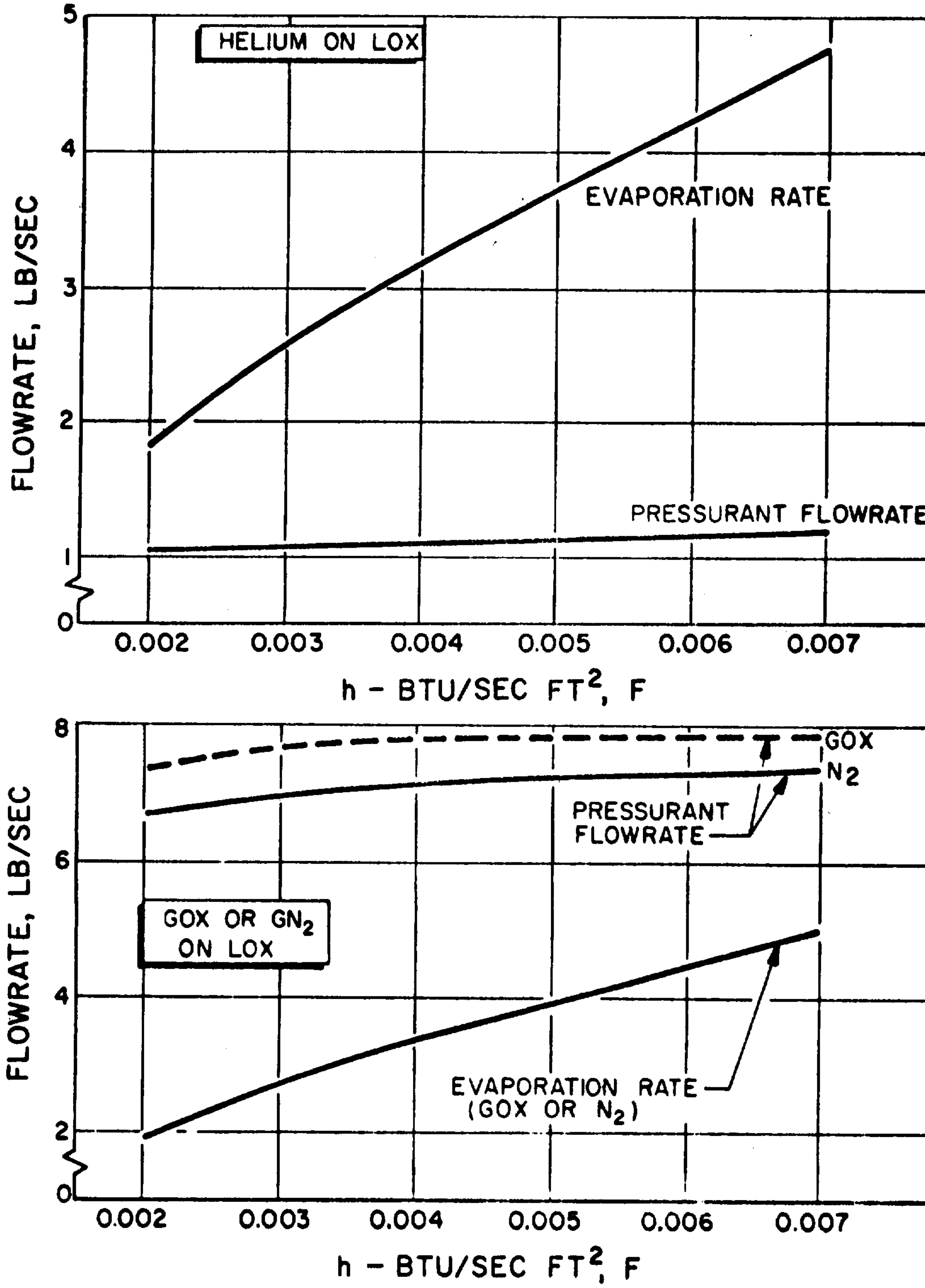


Figure 16. Effect of h on Required Pressurant and Evaporation Flowrates for F-1 LOX Tank
Note: Pressurant Gas Temperature, 960 R

R = gas constant, ft-lb/lb R

T = temperature, R

$\dot{V}$ = volume flowrate, ft^3/sec

$\dot{W}$ = flowrate, lb/sec

<u>Subscripts</u>

G = pressurizing gas

EV = evaporated propellant

T = total

u = ullage

S = saturation

L = liquid

APPENDIX D

WEIGHT ESTIMATES

STORAGE BOTTLES

Required pressurant storage tank volumes were determined as a function of the amounts of the various pressurants required during flight and the initial and final conditions existing in the storage containers. For the systems utilizing near liquid helium, it was assumed that 20 percent of the initial stored amount of pressurant would be residual at system shut-down, and that initial and final storage tank pressure would be 400 and 300 psia respectively. For the gaseous helium systems, initial and final storage sphere pressures were taken as 3250 psia and 350 psia respectively. Isentropic expansion of the gas in the bottle was assumed to determine the final gas temperature and residual gas weight. In the nitrogen and hydrogen systems the liquid pressurant was assumed to be completely expelled by the auxiliary pressurant gas at a pressure of 350 psia. Required total storage tank volumes were then found as:

$$V_t = \frac{W_u + W_r}{\rho_o} \qquad \text{(nomenclature, page 60)} \qquad (1)$$

The use of titanium as a storage bottle material was found to be practical in all systems except those involving near-liquid helium. In these systems, the weight advantage of titanium over Inconel-X as an inner shell material was not felt to be of sufficient magnitude to warrant the use of high cost titanium and the possible trouble areas associated with building large diameter titanium tanks with complicated internal hardware to operate at near-liquid helium temperature. Thus, bottle weights were calculated assuming titanium alloy as the inner sphere material in systems 2 and 4 through 8, and Inconel-X as the inner sphere material in systems 1 and 3. Strengths and densities of these two materials were taken as:

	S_y	S_u	ρ_m
Titanium	Not Used	165,000 psi	0.16 lb/in^3
Inconel-X	120,000 psi	Not Used	0.30

Wall thicknesses of the storage spheres were computed from the formula:

$$t = \frac{F_s P_o D}{4S} \qquad (2)$$

The value of the safety factor was taken as 1.50 for Inconel-X (based on yield strength) and 2.0 for titanium (based on ultimate strength).

The inner sphere weight was computed as:

$$W = \rho_m A t \qquad (3)$$

To account for mounting brackets, etc., an additional 10 percent of the inner sphere weight was included in the final weight values. Also included, where applicable, were the weights of insulation and outer shells, internal heat exchangers, and other required hardware, including control transducers, thermal equalizers, etc.

Fiberglass wrapped tanks were also considered but were ruled out for the present due to the small weight savings over titanium tanks and the fact that their use at cryogenic temperatures has not been fully investigated. Ultimate strength of fiberglass wrapping is, however, increasing rapidly as experience is gained in this field. Fiberglass is increasingly attractive as a product improvement item under the F-1 contract. Some information on fiberglass is listed in the following table:

	* Ultimate Strength-psi	Density lb/in^3	Cost
1960 Predicted for 1963	160,000 325,000	0.076 0.076	About 1/10 that of titanium.

* For Ovaloid Tanks

In addition to the preceding study, an investigation was made as to the relative advantages and disadvantages of using multiple storage tanks. These are listed below:

<u>Advantages</u>

1. More convenient size for handling and installation in the missile.
2. Pressurant storage capacity is easily varied
3. More practical size when considering diameter restriction imposed by the state of the art of manufacture of titanium pressure vessels

<u>Disadvantages</u>

1. Plumbing complexity
2. Cost

3. Controls complexity in the case of the "near liquid" helium systems

4. Increase in insulation weight of "near liquid" helium systems

5. Increased heat leak of "near liquid" helium storage tanks

In view of the above, it is felt that system 2, 4, 5, 6, 7 and 8 should be designed using multiple tanks. Use of multiple bottles for the "near liquid" helium systems 1 and 3 is not felt to be practical in view of the decreased storage life and the increased controls problems.

Tank Nomenclature

V_t = storage tank volume, ft^3

W_u = pressurant weight required in missile tank, lb

W_r = pressurant residual weight, lb

ρ_o = initial pressurant density in storage sphere, lb/ft^3

S_u = ultimate tensile strength, psi

S_y = yield tensile strength, psi

ρ_m = material density, lb/in^3

S = tensile strength, psi

P = operating pressure, psi

D = tank diameter, in.

F_s = safety factor

W = weight of storage inner sphere, lb

A = storage sphere surface area, in^2

t = storage sphere wall thickness, in.

HEAT EXCHANGERS

Main Engine Heat Exchangers

The main engine heat exchanger weight was estimated as follows:

1. Knowing the required pressurant flowrate and exchanger outlet temperature, the exchanger UA was calculated from:

   $$UA = q/\Delta T_1 \qquad \text{(nomenclature, page 63)} \qquad (4)$$

2. The type of heat exchanger was chosen on the following basis:

Type Unit	P_c	Design	* U	** Nom. ΔP_c
a. Vaporizing section	--	Bare coiled tube	70	35
b. Gas to gas and super-heater	>100	Extended surface	55	150
	<100	Tube bundle	55	15

* Based on past experience using LOX/RP-1 combustion products as the heating medium.

** These figures used in pressure loss calculations.

3. With the value of U from the above table, the required surface area was obtained from item 1. The heat exchanger weight was then estimated by means of the following empirical equations. (Based on construction using a metal with a 40,000 psi yield at the operating temperature.)

a. Bare Coiled Tube

$$\left.\begin{array}{l} W_c = 0.014\ P_c A \\ W_s = 0.014\ P_s A \\ W_m = 0.5\ A \end{array}\right\} W_t = W_c + W_s + W_m$$

b. Tube Bundle $W_t = 0.0025\ P_c A$

c. Extended Surface $W_t = 0.0017\ P_c A$

4. Hot gas side pressure drop was assumed to be 15 psi in all cases.

Storage Tank Heat Exchangers

Internal tank heat exchanger coefficients were estimated from the conventional natural convection equation for horizontal cylinders (Ref. A3).

$$Nu_o = 0.11 \left[Gr \cdot Pr \right]_o^{1/3}$$

Using this as the controlling resistance, the exchanger surface area and weight were easily calculable.

[redacted]

Heat Exchanger Nomenclature

A = required surface area, ft^2

P = operating pressure, psia

ΔP = allowable cold side pressure loss, psi

q = heat transfer rate required to bring cold side fluid to desired outlet condition, Btu/hr

ΔT_l = log mean temperature difference between hot and cold side fluids, R

U = over-all coefficient of heat transfer, Btu/hr ft^2 F

Nu = Nusselt number

Gr = Grashof number

Pr = Prandtl number

Subscripts

c = tube or coil

s = shell

m = miscellaneous

t = total

o = outside

LINES AND FITTINGS

Line sizes and weights were estimated as follows:

1. The line length was estimated from the assumed tank configuration and present engine drawings.
2. The minimum allowable pressure at the main tank regulator was assumed to be 200 psi and the minimum supply pressure 350 psi. (Except in oxygen systems where pump discharge pressure was assumed).
3. The allowable line pressure loss is then

$$\Delta P_L = P_b - P_L - \Delta P_m$$ (nomenclature, page 65)

4. The line diameter was then estimated from

$$D = \sqrt[5]{\frac{8\,\dot{w}^2\,fL}{\pi^2 g\,\rho_{av}\,\Delta P_L}}$$

5. The wall thickness was calculated from

$$t = \frac{F_s\,P_o\,D}{2S}$$

6. The weight is then given by

$$W_L = \pi DLt\,\rho_m$$

Line and Fitting Nomenclature

P_L = allowable line loss, psf

P_m = heat exchanger pressure loss, psf

P_b = minimum supply pressure, psfa

P_L = minimum allowable pressure at regulator, psfa

P_o = maximum operating pressure, psfa

$\dot{w}$ = pressurant flowrate increased by 50 percent for transients, lb/sec

f = Darcy friction factor (increased 50 percent for fittings, etc.)

L = line length, ft

D = line diameter, ft

g = 32.2 ft/sec^2

ρ_{av} = average fluid density in line, lb/ft^3

S = yield or ultimate stress as applicable, psf

F_s = safety factor (1.5 based on yield - 2.0 based on ultimate)

t = wall thickness, ft

ρ_m = metal density, lb/ft^3

W_L = line and fitting weight, lb

Miscellaneous

The weights of the hoses, relief valves, solenoid valves, check valves, and orifices were obtained from Fig. 17 and 18, which represent average values based on existing designs.

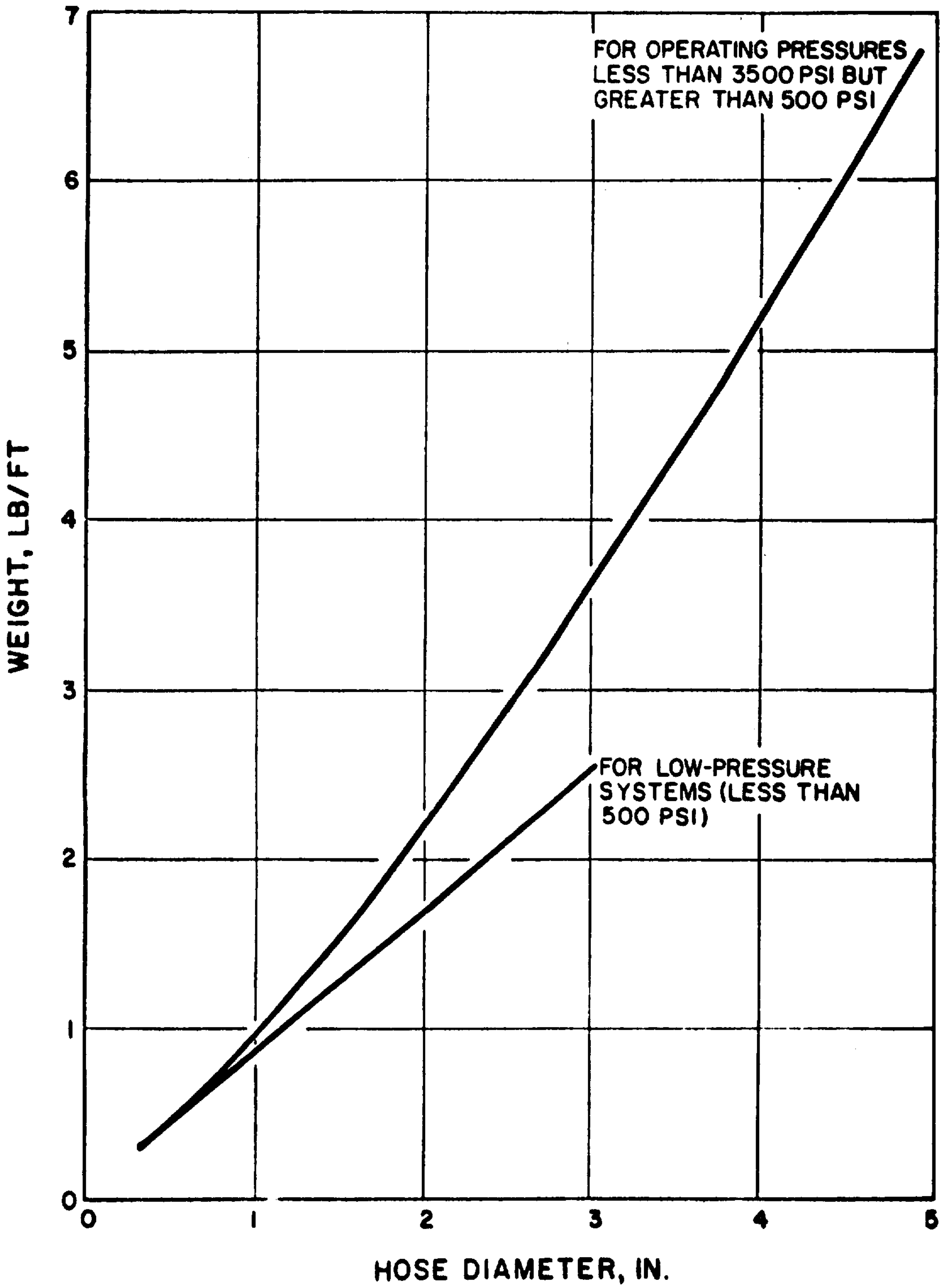


Figure 17. Weight of Flex Hoses vs Hose Diameter

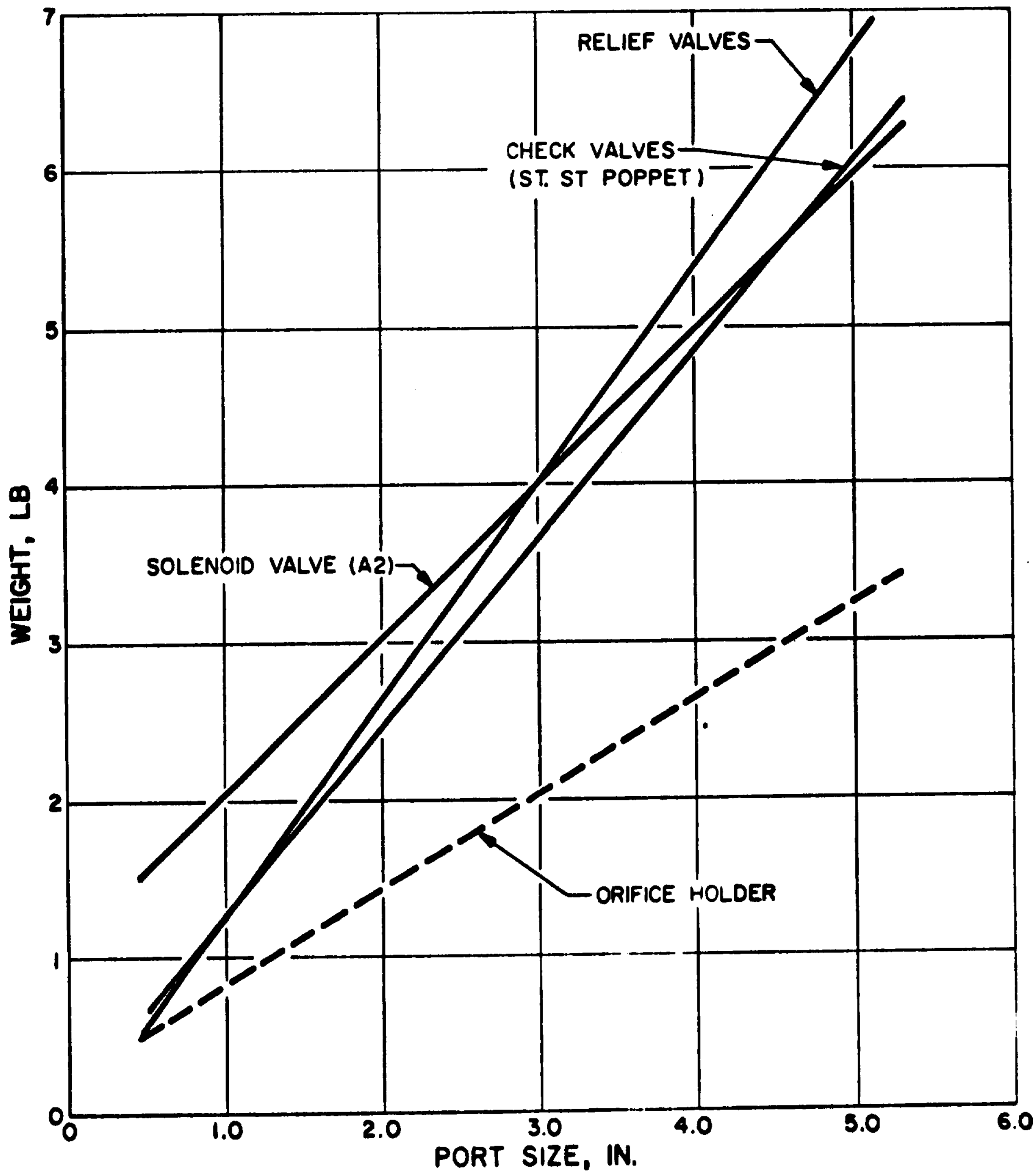


Figure 18. Weight of Control Valves vs Port Size

REGULATOR WEIGHT

The weight of the tank pressure regulators was assumed to be a function of regulator inlet valve size, and was obtained from Fig. 19, which is based on the weights of regulators designed by Rocketdyne in the past. Regulator inlet valve size was computed from

$$d = \left[\frac{1.5\ \dot{W}\ RT_g}{0.785\ CP_rS} \right]^{1/2}$$

Where an allowance of 50 percent has been made for transients and

d = inlet valve diameter, in.

$\dot{W}$ = pressurant flowrate, lb/sec

T_g = inlet temperature, R

C = coefficient of discharge of inlet valve, dimensionless

R = specific gas constant, ft/R

P_r = minimum inlet pressure, psia

S = 4.11 ft^3/sec for helium

3.88 ft^3/sec for nitrogen, oxygen, and hydrogen

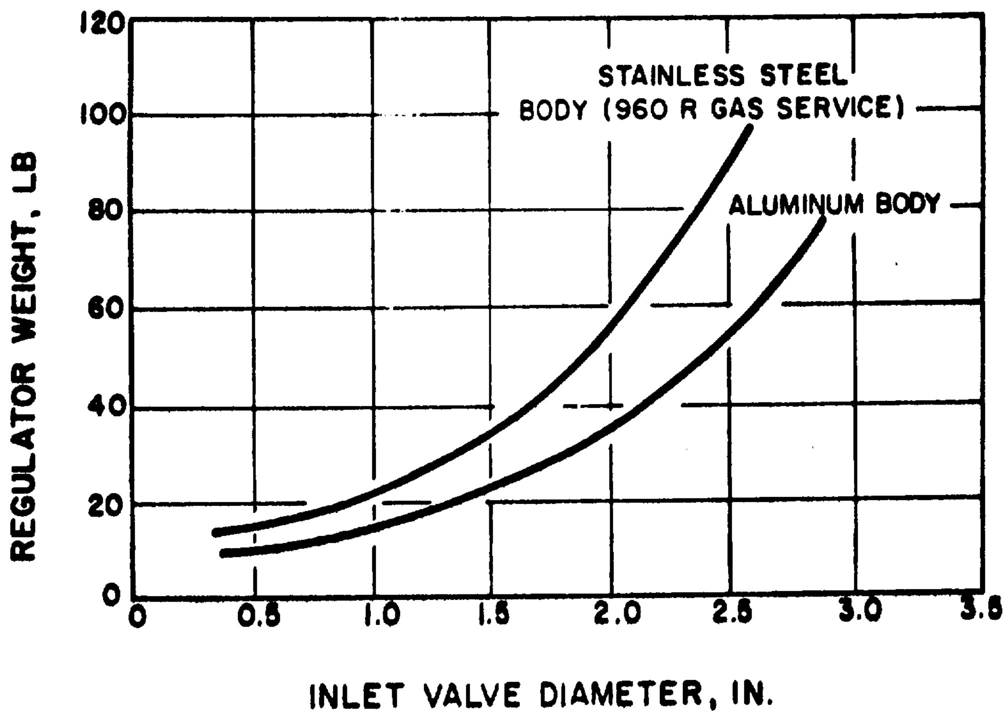


Figure 19. Tank Pressure Regulator Weight vs Inlet Valve Diameter

APPENDIX E

STATE OF THE ART

A considerable portion of the study effort was devoted to investigation of the relative state of the art associated with the subject pressurization systems. System cost and reliability are greatly dependent on the state of current technology, and these figures reflect this dependence.

The experience of Rocketdyne and other organizations in the development of tank pressurizing equipment and the handling of liquefied gases was drawn upon in arriving at the state-of-the-art estimate. The estimate is based on the assignment of one of the following "levels" of state of the art to each system component:

Excellent	=	Routine technique, successfully used in the rocket engine and missile fields.
Good	=	Extrapolation of a routine technique requiring minimum development.
Fair	=	Reasonable knowledge of the problems to be encountered, but requiring considerable development effort.
Poor	=	Involving techniques or components for which there is little or no past experience, and minimum ability to predict problem areas, therefore requiring lengthy development.

The assignment of these levels of state of the art is shown in Table 4. Only the major components and the pressurant handling techniques are presented. Those components not covered are common to all systems and do not affect their relative standing.

The technology of building flight-weight, pressurant-storage spheres was felt to be good in all cases except those involving liquid helium. The requirement of providing insulation and vacuum jacketing for a flight tank, as well as the required tank transducers and internal hardware, reduced the rating of the tank for System 1 to fair. The complication of compatibility resulted in a score of fair-minus for the tank of System 3.

Main heat exchangers were given a rating of excellent except for the heat exchanger of System 3, where the compatibility requirement reduced its rating to excellent-minus.

Low-pressure flex lines were rated excellent, while the high-pressure flex lines of Systems 4, 5, and 8 were rated good.

Static controls were rated good except in the systems involving liquid helium or hydrogen as pressurant, where the relative newness of these fluids reduced the ratings to fair and fair-plus respectively for Systems 1 and 7. Static controls for System 3 were rated fair-minus due to the compatibility requirement. The dynamic controls for nitrogen and oxygen were rated good based on past experience. For those systems involving helium and hydrogen as pressurants, the dynamic controls were rated fair-minus to fair-plus, as a function of amount of past experience for the particular application.

TABLE 4

STATE OF THE ART

	System Number and Description							
	1	2	3	4	5	6	7	8
Category	Optimum LHe	Optimum LN_2	Compatible LHe- LN_2	Gaseous He with LN_2 Cooling	Gaseous He with LH_2 Cooling	GOX on LOX LN_2 on Fuel	GOX on LOX LH_2 on Fuel	GOX on LOX Gaseous He on Fuel (LN_2 Cooling)
Pressurant Tanks	F	G	F-	G	G	G	G	G
Heat Exchangers	E	E	E-	E	E	E	E	E
Flex Lines	E	E	E	G	G	E	E	G
Static Controls	F	G	F-	G	G	G	F+	G
Dynamic Controls	F	G	F-	F	F	G	F+	F
Pressurant Handling Techniques	P	E	P	G	G-	E	G-	G
System Score	F+	E-	F	G	G	E-	G	G

Key:

E: Excellent

G: Good

F: Fair

P: Poor

Pressurant handling techniques for oxygen and nitrogen were considered excellent. Systems 4 and 8, utilizing LN_2-jacketed high-pressure helium, were rated good due to the added complication of providing the coolant. For Systems 5 and 7 the problem of handling LH_2 resulted in a rating of good-minus. Handling of LH_2 is becoming relatively routine, although requiring safety equipment and procedures due to the fire hazard.

Handling of liquid helium was given a state-of-the-art rating of poor. Experience in the handling and transfer of large quantities of this fluid is virtually nonexistent. The problems associated with the handling and storage of liquid helium warrant further consideration and are discussed separately in Appendix H.

An over-all system state-of-the-art rating was determined for each system, based on the above individual ratings. To accomplish this, numerical equivalents were assigned to the ratings ranging from zero for a rating of poor-minus to a value of ten for excellent. It was felt that the importance of the pressurant handling category warranted assignment of double value to its state-of-the-art rating. The numerical scores were then totaled and averaged, resulting in the system state-of-the-art ratings as shown in the bottom line of Table 4.

APPENDIX F

RELIABILITY

The relative reliability of each of the eight pressurization systems under study, after a 30-month development period, was determined by multiplying together the estimated reliabilities of the individual components of the system. Component reliabilities were conservatively estimated on the basis of Rocketdyne experience in the development and production of similar hardware items on other contracts.

The method outlined in the preceding paragraph yielded the following values of the relative reliabilities of the eight systems:

System No.	Relative System Reliability, percent
1	94.9
2	98.4
3	94.6
4	98.0
5	97.6
6	98.3
7	96.7
8	98.2

Because of the method by which these relative reliability values were obtained, it is believed that they are conservative and that the reliability which would actually be attained in the development of any one of the systems would be considerably higher than the above figures indicate.

APPENDIX G

COST ESTIMATES

Cost estimates were prepared for comparative purposes only and are not to be considered as firm estimates of the cost of any one system. The following items are included in the estimates:

1. Cost of additional facilities required for the handling and storage of helium and liquid hydrogen
2. Cost of pressurant required for the development of the systems and the system components
3. Cost of ground support equipment (GSE)
4. Cost of system hardware
5. Cost of pressurant needed for operation of the system

No costs were estimated for System 5 (high-pressure gaseous helium with liquid hydrogen cooling), because its low reliability and high weight relative to System 4 (high-pressure gaseous helium with liquid nitrogen cooling) were deemed to rule it out of consideration.

Estimated component hardware costs are summarized in Table 5. For the purpose of this estimate each system was assumed to consist of the items of hardware shown in the system schematics, Fig. 1 to 6, inclusive.

TABLE 5

HARDWARE COSTS

(When purchased in small quantities)

System Number	Cost, Thousands of Dollars							
	1	2	3	4	5	6	7	8
Storage Tanks	40.0	30.0	50.0	40.0	Not Estimated	25.0	36.0	30.0
Heat Exchangers	7.0	7.0	9.0	8.0		8.0	8.0	8.0
Regulators	16.0	16.0	16.0	16.0		16.0	16.0	16.0
Bypass Valve	3.0	--	3.0	--		--	3.0	--
LN_2 Tank Pressure Regulator	--	4.0	--	--		4.0	4.0	--
Flex Lines	2.5	2.5	2.5	3.5		2.5	2.5	4.5
Disconnects	3.0	2.0	3.0	2.0		2.0	2.5	2.0
Relief Valves*	11.0	12.0	11.0	11.4		12.0	11.0	11.0
Shut off Valves	1.6	1.2	1.6	3.2		1.2	1.4	3.2
Check Valves	1.2	1.8	1.2	1.2		0.6	0.6	0.6
Misc. (Lines, Fittings, etc.)	2.0	2.0	2.0	2.0		2.0	2.0	2.0
Total	87.3	78.5	99.3	87.3		73.3	87.0	77.3

* Including main tank relief valves.

Estimated costs of facility additions, ground support equipment and pressurant are shown in Table 6. Pressurant costs are based on the following unit prices:

Liquid Helium	$10.90 per lb F.O.B. Amarillo, Texas
Liquid Oxygen	$ 0.13 per lb delivered
Liquid Nitrogen	$ 0.09 per lb delivered
Liquid Hydrogen	$ 0.55 per lb delivered
Helium Gas	$ 2.50 per lb delivered

Over a period of time the total cost of a system will be the initial cost plus the operating costs. As a first approach, operating costs were assumed to consist of:

1. Basic manufacturing cost of hardware
2. Cost of pressurants required for system operation

Other costs will be incurred, such as cost of installing the system in the vehicle, field service support, spare parts, maintenance, inspections and checkouts, manpower to operate the system, etc. These additional costs were considered too nebulous to estimate at this time.

The initial cost was assumed to be the added development cost of the system (Table 6), plus cost of one set of GSE, plus cost of one set of hardware, plus pressurant cost for 8 captive tests of the first vehicle to check out the pressurization system (and all other systems). Cost vs time for four launches per year of a vehicle powered with a single F-1 engine is shown in Fig. 20, on the assumption that each vehicle undergoes one static test prior to flight.

TABLE 6

ESTIMATED COSTS OF FACILITY ADDITIONS, GSE AND PRESSURANT

System Number	Cost, Thousands of Dollars						
	1	2	3	4	6	7	8
Additional Facilities							
Equipment	402.0	0.0	402.0	107.0	0.0	101.0	107.0
Installation	12.0	0.0	12.0	15.0	0.0	15.0	15.0
Subtotal	414.0	0.0	414.0	122.0	0.0	116.0	122.0
Development Pressurant	1035.0	46.0	1526.0	244.0	50.0	62.0	178.0
Total Added Development	1449.0	46.0	1940.0	366.0	50.0	178.0	300.0
Total Added Development, Relative to System 2	1403.0	0.0	1894.0	320.0	4.0	132.0	254.0
Ground Support Equipment	392.5	139.5	465.0	295.0	119.5	145.0	215.0
Operational Pressurant (Per Run)	5.5	0.4	7.6* 0.2**	1.3	0.4	0.3	1.0

* When using LHe

**When using LN_2

NOTE: Costs of System 5 were not estimated. (see page 77).

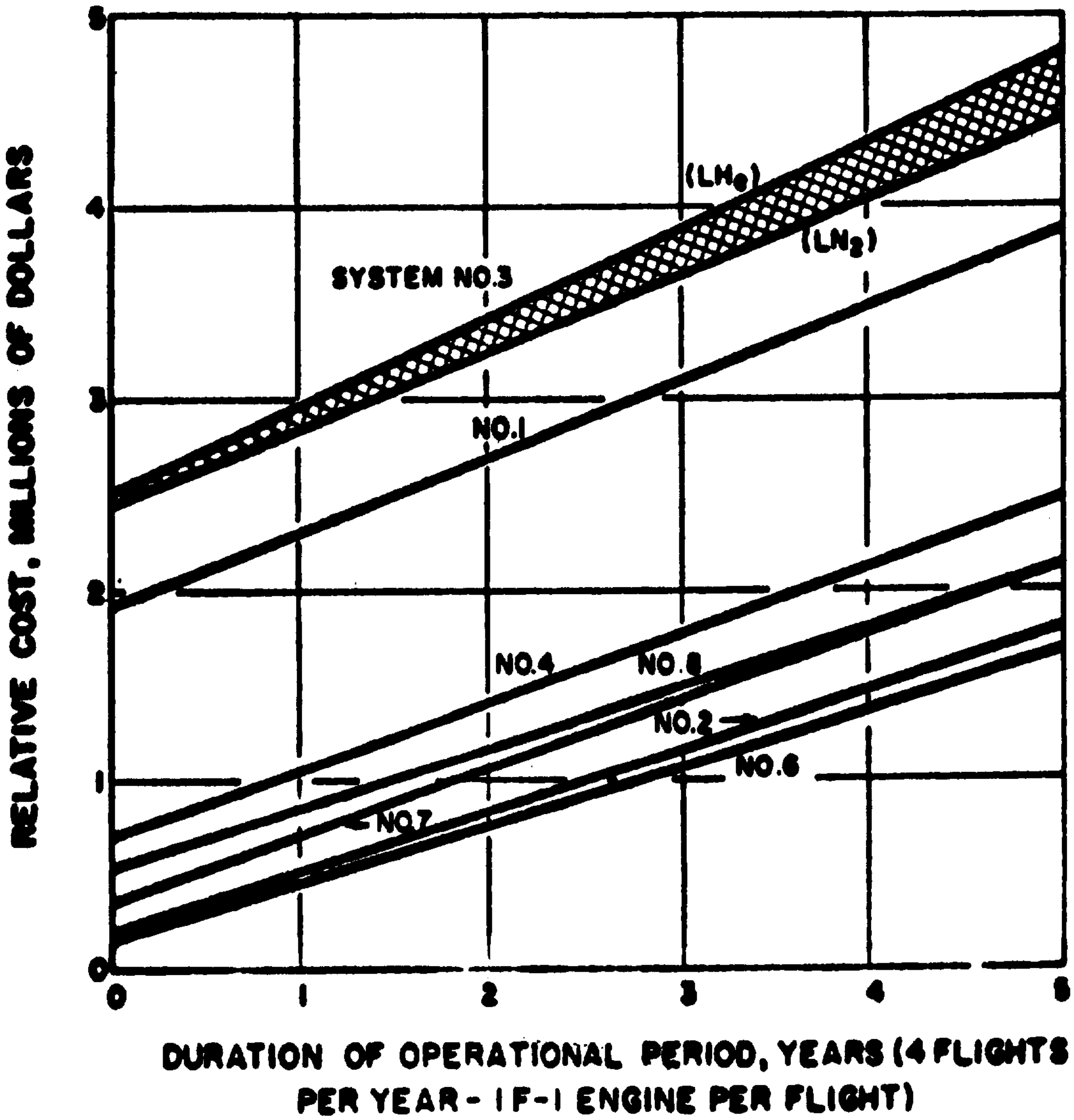


Figure 20. Relative Cost vs Duration of Operational Period

Calculations were also made for an 8-engine vehicle, and up to 10 years of operation. There were no cases in which the investment in a system of higher initial cost, but lower operating cost, would be repaid within a reasonable period.

The hardware cost for System 3 (LHe compatible LN_2) exceeds that of System 1 (optimum LHe) by $12,000. The saving in pressurant cost by operating System 3 with LN_2 is $4,300 per firing, but when System 3 is operated with LHe it costs $2,100 more than if System 1 were used. Therefore, it is impossible to save any money by using System 3 unless a very large portion of all firings are made with LN_2, and unless at least three firings are made with each production pressurization system. Even if these conditions were fulfilled, amortization of the initial cost of System 3 as compared with System 1 could not be achieved within any reasonable period of time. Therefore System 3 cannot be justified in preference to System 1 on economic grounds.

APPENDIX H

LIQUID HELIUM STORAGE AND HANDLING

The successful use of liquid helium as a tank pressurant depends upon the solution of the numerous problems involved in the storage and handling of the fluid in large quantities. Because of the high cost ($10.90 per lb in large quantities), every effort must be made to reduce losses due to external heat leak and "flash off" during transfer operations. In addition it is understood that a bill is now pending in Congress which, if passed, will require the recovery of helium wherever possible. This will result in a costly and time-consuming operation at an already complicated launch site.

The following discussion serves only to point out some of these problem areas and to indicate those in which progress has been made.

Helium has the lowest boiling point of all the liquefied gases (7.5 R at ambient pressure). Long-term storage and transfer operations must be carried out in pre-chilled, highly insulated and costly "Dewars", since its heat of vaporization is exceedingly low (9 Btu/lb at ambient pressure compared with 90, 85 and 190 Btu/lb for LOX, LN_2 and LH_2 respectively).

Extensive research by various manufacturers has resulted in the development of the so-called "super" insulations (e.g., Linde SI-4) which are in effect laminated radiation shields. These are, however, very delicate to handle and must be used in a vacuum of 10^{-5} mm of Hg to be fully efficient. The attainment of high vacuums is in itself no small problem (due to outgassing of commercially available metals, etc.).

The low critical pressure (33.2 psia) and temperature (10 R) put a severe restriction on the pressure levels available for transferring the liquid between Dewars. Since the fluid must be in the saturated condition upon entering the final container (to obtain minimum loss), accurate estimates of line pressure drop must be made. This is extremely difficult since even small amounts of evaporated fluid result in enormous pressure loss changes, and, of all flow phenomena, the prediction of two-phase losses is least understood.

Advances made to minimize these problems are:

1. Liquefied helium is available in quantity from the Navy liquefaction plant (70 liters per hr capacity) at Lakehurst, New Jersey.
2. Transport Dewars are commercially available which limit the loss to about 0.1 percent per day.
3. Small liquid helium pumps have been built.
4. The National Bureau of Standards at Boulder, Colorado now feels that the fluid may be successfully transferred through relatively long piping systems (perhaps up to 100 ft).
5. Proposals have been received on flight-weight Dewars for the originally proposed F-1 pressurization system and have been reviewed by NBS and regarded as feasible, although NBS considered the technical proposals optimistic in certain respects (Ref. C1).

6. Recovery systems have been built and operated successfully by several companies (such as Wyle Laboratories) although not on the scale anticipated for the F-1 program.

The use of liquid helium in the F-1 vehicle thus presents no insurmountable problems but should still be viewed with caution. It is very likely that its use at the present time is unjustified on the basis of the above discussion alone.